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Part II, Volume II

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(UNCLASSIFIED TITLE)

Environmental Control Systems Selection for Manned Space Vehicles

Volume II: Appendix I -- Missions, Vehicles, and Equipment

TECHNICAL REPORT NO. ASD-TR-61-240, PT. II, VOL. II

October 1962

Flight Accessories Laboratory
Aeronautical Systems Division
Air Force Systems Command
Wright-Patterson Air Force Base, Ohio

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NOTICES

FOREWORD

This appendix to "Environmental Control Systems Selection for Manned Space Vehicles" has been separated from the main volume and classified principally because of the possibility of suggestion or revealing portions of Air Force planning programs or underlying concepts. This report, as well as the main report to which it is appended, is one of a series on space vehicle thermal and atmospheric control systems.

This report is classified SECRET because it describes an offensive weapon system.

ABSTRACT

This Abstract is Unclassified

Four versions of manned orbital reentry basepoint vehicles are developed for the purpose of providing reference points for determination of the thermal and atmospheric control requirements of realistic vehicles. Two of the vehicles (i.e., Vehicles 1A and 1B) were developed in Phase I of this study series (ASD Technical Report 61-240, Pt I) and will only be summarized here. The remaining two vehicles (Ballistic Reentry and Lenticular Reentry) are presented in greater detail in this report.

In addition to the development of specific vehicles, general data have been compiled on the more important aspects of manned space vehicle design, (i.e., flight vehicle power, structures, effects of meteoroids, mission equipment, and examination of these general data for environmental requirements.

PUBLICATION REVIEW

This report has been reviewed and is approved.

FOR THE COMMANDER:

Frederick W. Thompson
WILLIAM C. SAVAGE
Chief, Environmental Branch
Flight Accessories Laboratory

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ILLUSTRATIONS

TABLES

LIST OF SYMBOLS

A	-	area	ft^2
C_p	-	drag coefficient	--
C_L	-	lift coefficient	--
L	-	lift	lbs
D	-	drag	lbs
d	-	diameter	ft
h_w	-	wall enthalpy	Btu/lb
h_s	-	stagnation enthalpy	Btu/lb
h_{eff}	-	effective heat of ablation	Btu/lb
$\dot{q}_o$	-	stagnation point heating rate	Btu/ft^2 sec
s	-	ablator thickness	ft
T	-	temperature	°R
t	-	time	seconds
V	-	vehicle velocity	ft/sec
W	-	weight	lbs
γ_e	-	flight path angle	degrees
ρ_∞	-	free stream (ambient) density	lbs sec^2/ft^4

Appendix I

MISSIONS, VEHICLES, AND EQUIPMENT

As stated in Section II of Volume I, the purpose of developing specific hypothetical vehicles is to provide tangible reference points for determination of the thermal and atmospheric control system requirements of realistic manned space vehicles. These vehicles serve as a means for: (1) identification of environmental factors such as cabin heat rejection, solar and aerodynamic heating, cabin pressure losses, and cabin atmospheric contamination; (2) establishment of environmental requirements of crew and equipment; (3) integration of thermal and atmospheric control systems into realistic vehicles; and (4) development of trade data useful in selecting and sizing thermal and atmospheric control systems.

The scope of this portion of the study is limited to the time period 1965 to 1975. Thus, projects Mercury and Dyna Soar are considered to be pre-1965, while planetary entry and landing missions are considered to be post-1975 (References 1 and 2).

Primary emphasis to date has been placed on the manned orbital reentry vehicle whose mission would be global surveillance and/or bombardment. This has been done inasmuch as such a vehicle is probably of the greatest immediate military interest and inasmuch as such a vehicle also serves as an excellent model for thermal and atmospheric control system design studies. Four variations of this orbital reentry vehicle were developed to establish the influence of crew size, mission duration, mission equipment, and flight vehicle power on thermal and atmospheric control systems. These three classes of the manned reentry vehicles were developed in detail sufficient to accomplish the purposes stated at the beginning of this appendix.

For convenience, the three subclasses of the manned orbital reentry vehicles have been designated as follows:

Vehicle 1A -- Five-man, 6-week, full-surveillance version

Vehicle 1B -- Two-man, 1-week, full-surveillance version

Vehicle 2A -- Five-man, 6-week, full-surveillance version

Vehicle 3A -- Four-man, 6-week, bombardment version

Manuscript released by authors 13 July 1962 for publication as an ASD Technical Report.

All four versions are boosted, orbital configurations with pilot controlled reentry trajectory. Vehicles 1A and 1B are winged, Vehicle 2A, ballistic, and Vehicle 3A, lenticular shaped. Vehicles 1A and 1B are briefly outlined here. A detailed description will be found in Reference 3. Vehicles 2A and 3A are discussed in more detail in this section of the appendix.

VEHICLE 1A

Vehicle 1A is a five-man, 6-week version having a complete complement of equipment for global surveillance. The configuration and many of the design details of Vehicle 1A are shown in Figure 1. The basic characteristics of this version are as follows:

Crew size	5
Gross launch weight	57,825 lb
Reentry weight	52,000 lb
Wing area	1325 ft^2
Reentry wing loading	39.2 lb/ft^2
Fuselage overall length (w/o flight vehicle power unit)	86-1/2 ft
Fuselage diameter (maximum inside)	8-1/4 ft
Total volume (separate on-duty, off-duty, and equipment compartments)	1500 ft^3
Equipment and crew heat rejection	20 kw (continuous and nearly steady)
Flight vehicle power	Nuclear
Sweep	73 deg
Leading edge radius	6 in.
Nose radius	12 in.

An alternate power unit, a solar turboelectric ("Sunflower") system, is shown in Figure 2. This system consists of a 50-foot-diameter erectable parabolic reflector, the turbine and working fluid, heat storage (possibly LiH), orientation system, and associated structure.

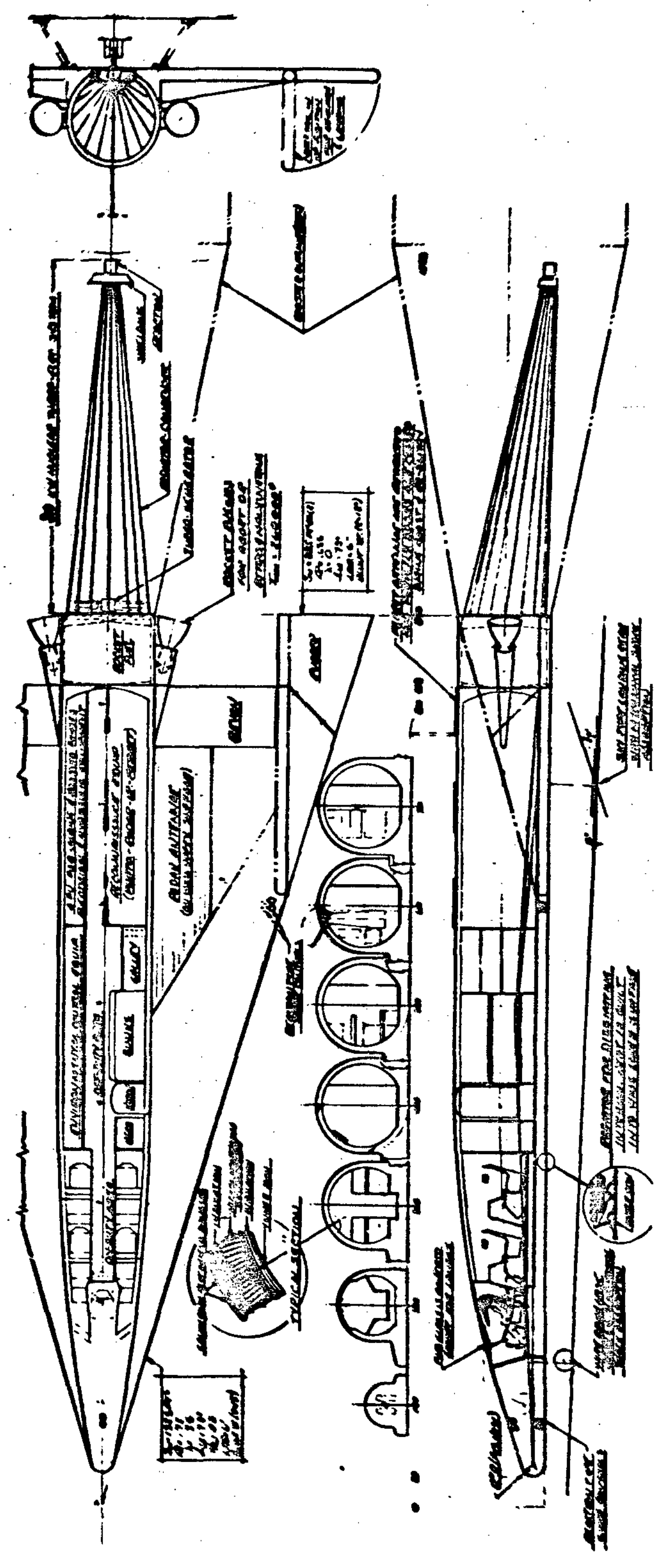


Figure 1 Vehicle 1A

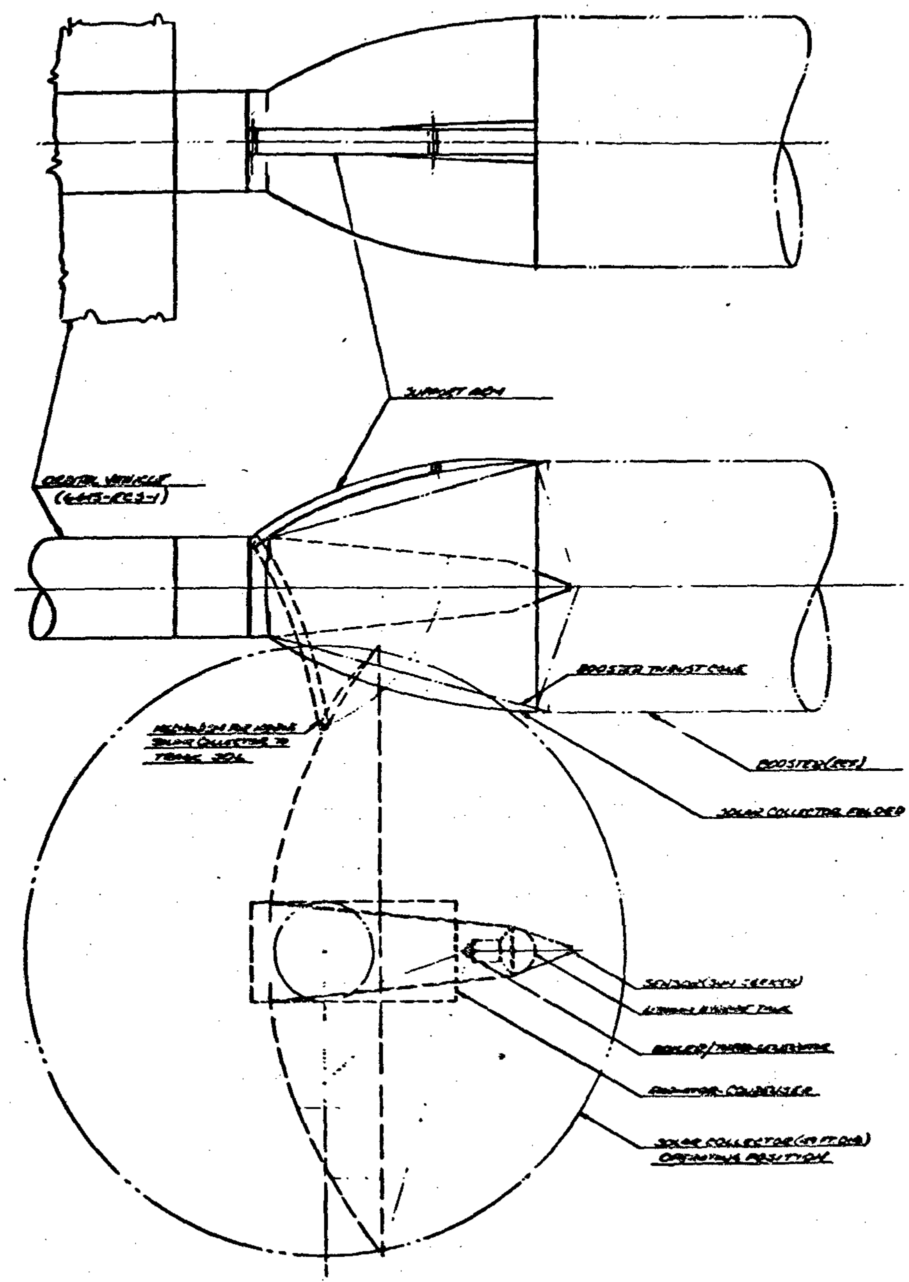


Figure 2 Alternate Flight Vehicle Power Systems for Vehicle 1A

A weight summary for Vehicle 1A is presented on Table 1. The total power requirement for the orbital phase is as follows:

Peak equipment requirements:	14.1 kw
Crew requirements (100-200 watts/man):	1.0 kw
Refrigeration:	5.0 kw
	20.1 kw

Reentry power requirements for Vehicle 1A are tabulated below:

Function	Duration (sec)	Power (kw)	Energy (kw-hr)
Comm & nav Environmental control Refrigeration for above	0.5150	2.7	3.8
Aerodynamic control	700-5100	40 (normal peaks)	24.3
		74 (unusual peaks)	22.6
		11 (minimum)	3.5
Approach and landing	5100-5150	58	0.8
			55.0

TABLE 1

WEIGHT SUMMARY FOR VEHICLE 1A

Structure:		26,300 lb
Wing	11,500	
Vertical tail	2,700	
Fuselage	7,950	
Landing gear	1,500	
Engine section	150	
Surface controls	2,500	
Power plant:		6,380 lb
Rockets	2,750	
Rocket controls	100	
Fuel system	1,000	
Flight vehicle power System (nuclear)	2,200	
Reentry APU	330	
Fixed equipment:		16,750 lb
Instruments	500	
Hydraulics	500	
Electrical	750	
Electronics (6400 plus structure)	8,000	
Furnishings	1,750	
T & A system	4,550	
Food and H_2O		
Data return capsules	600	
Auxiliary gear	100	
Weight empty		49,430 lb
Crew, fuel, and other:		8,395 lb
Crew (5 men)	1,500	
Fuel (abort/maneuver & retro)	5,470	
Trapped fuel	400	
Fuel (attitude control)	500	
Oil	25	
Miscellaneous	500	
Launch gross weight		57,825 lb
Orbital weight (2770 for maneuver)		55,050 (if all maneuvering fuel used)
Reentry weight (2700 for retro, 350 for attitude control)		52,000 lb

VEHICLE 1B

The size of the basic vehicle and the duration of the mission have been varied in order to illustrate possible crossover points for changes in flight vehicle power systems, environmental control systems and other subsystems. The second version has purposely been kept similar in configuration in order to facilitate comparison of any critical parameters. Accordingly, in what follows only the difference between Vehicle 1B and Vehicle 1A is summarized.

Vehicle 1B is a two-man, 1-week version of the global surveillance vehicle as shown in Figure 3. A full complement of reconnaissance equipment is included. Characteristics of this version are as follows:

Crew size	2
Gross launch weight	46,430 lb
Reentry weight	42,000 lb
Wing area	915 ft
Reentry wing loading	46 lb/ft^2
Fuselage length (W/O flight vehicle power unit)	61-3/4 ft
Fuselage width (maximum inside)	8-1/4 ft
Fuselage height (maximum inside)	7-1/2 ft
Total volume (separable crew and equipment compartments)	1200 ft^3
Equipment and crew heat rejection	20 kw (continuous and nearly steady)
Flight vehicle power	Nuclear or solar turboelectric

The weight summary for Vehicle 1B is given in Table 2.

As mentioned previously, reference is made to Phase I of this study series for a detailed description of Vehicles 1A and 1B (Reference 3).

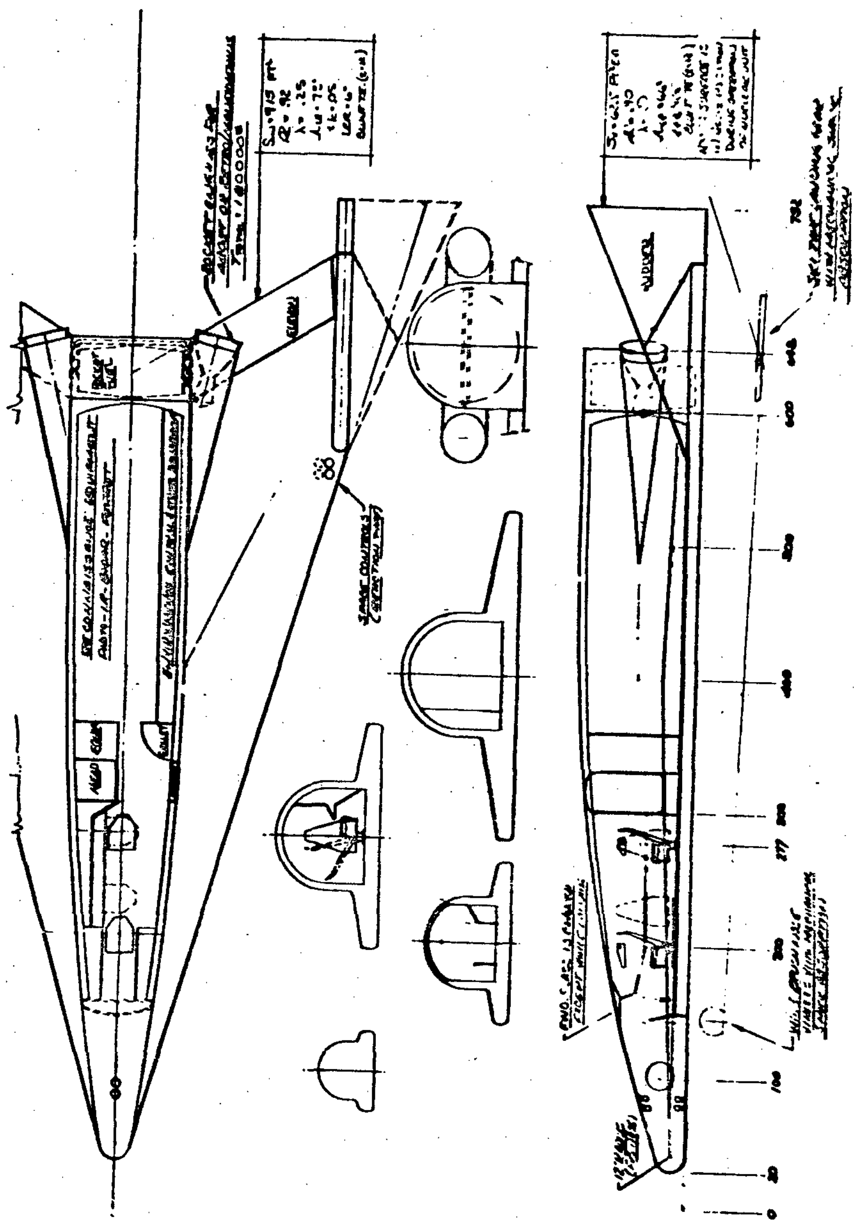


Figure 3. Vehicle 1B

TABLE 2

WEIGHT SUMMARY FOR VEHICLE 1B

Structure		20,050 lb
Wing	8000	
Vertical Tail	2000	
Fuselage	6500	
Landing Gear	1200	
Engine Section	150	
Surface Controls	2200	
Power Plant		5,600 lb
Engine	2000	
Engine Controls	100	
Fuel System	1000	
Flight Vehicle Power System (Nuclear)	2200	
Reentry APU	300	
Fixed Equipment		15,050 lb
Instruments	400	
Hydraulics	400	
Electrical	700	
Electronics (6400 plus automation plus structure	3500	
Furnishings	1000	
T & A System, Food & H_2O	3550	
Data Return Capsules	400	
Auxiliary Gear	100	
Weight Empty		40,700 lb

TABLE 2 (Con't)

WEIGHT SUMMARY FOR VEHICLE 1B

Crew, Fuel and Other		5,730 lb
Crew (2 Men)	600	
Fuel (Abort/Maneuver & Retro)	4380	
Trapped Fuel	350	
Fuel (Attitude Control)	175	
Oil	25	
Miscellaneous	200	
Launch Gross Weight		46,430 lb
Orbital Weight		44,200 lb
Reentry Weight		42,000 lb

BALLISTIC REENTRY VEHICLE

This section describes conceptual designs and characteristics of ballistic type manned vehicles as they pertain to the thermal and atmospheric (T and A) control subsystem. Major emphasis is on the delineation of a base point vehicle whose mission requirements are the most severe from the standpoint of crew size and mission equipment. Other vehicles to perform less severe missions are briefly described. A significant T and A consideration resulting from this vehicular study is that it appears necessary for the crew to leave the vehicle while it is in orbit to erect or at least to inspect or align certain pieces of large equipment, such as the solar collector and the side looking radar.

The mission for which the base point vehicle is designed is a six week earth orbital surveillance mission with a crew of five men to operate the electronic and photographic equipment and to fly the vehicle. References 4 and 5 furnished useful information on the vehicle external shape and the surveillance equipment required for the mission.

Configuration Selection

Earth Entry Considerations

The "ballistic" reentry vehicle is interpreted herein as meaning primarily a non-winged vehicle, and possibly blunt and symmetrical. The rigors of a pure ballistic entry (load factor of at least 8 g's) and the lack of atmospheric maneuvering capability suggest that a "semi-ballistic" or lifting body entry may be preferable. Various types of lifting bodies have been considered for entry vehicles, such as lifting cones, half-cones and variations thereof, and symmetric bodies with blunt faces that obtain lift by flying at what would normally be called a negative angle of attack. The latter type of vehicle can be considered an outgrowth of the Mercury type ballistic capsule and as such can be expected to have many of its characteristics proved during successful execution of the Mercury program. It is the semi-ballistics, symmetric, blunt faced vehicle which is chosen for the present study; a hypersonic L/D of about 0.5 with $\frac{W}{C_D A} = 50$ will be seen to provide a reasonable entry environment, the symmetric shape is advantageous from the standpoint of launch stack configuration, and the blunt face forward entry minimizes or eliminates crew repositioning for launch and entry, besides being more "conventional" as per other arguments above.

Orbital Considerations

To satisfy the 6 week full surveillance mission requirement with a crew of 5 men suggests that the orbital configuration should have large down and up-looking areas to accommodate the surveillance equipment and radiators and a volume sufficient to make life comfortable for the crew so they may perform their tasks with efficiency and alacrity. The problem is to provide the necessary area for orbital operation, given a blunt reentry shape which inherently has a low area to volume ratio.

Vehicle Evolution

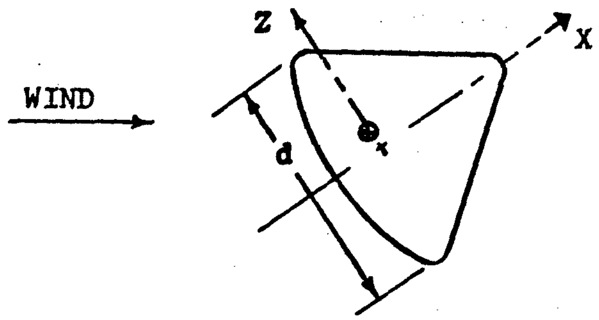
(a) Modular Arrangements

Mission and life support modules which are placed in orbit permanently have the advantage of not being required to withstand entry. An arrangement such as this would be serviced via a shuttle vehicle. This concept lacks system and operational flexibility, and maintenance and repair must be performed in orbit. Also, for emergency return of the crew from orbit, a reentry vehicle must always be at the station. Rendezvous capabilities must be provided in the shuttle vehicle.

A non-reusable modular concept would provide a very flexible operational system. Less ambitious missions can be accomplished with reduced system launch weight, employing modules tailored for the particular task. As above, only the reentry vehicle need be hardened for entry. However, it is desired to return the equipment so it may be reused.

(b) Integral Arrangements

As mentioned in the section on earth entry considerations, a variety of semi-ballistic base point vehicle shapes may be examined. Since this study emphasizes the T and A system with vehicular concepts entering only as they effect the T and A system, a comprehensive vehicle design trade-off is not in order. Rather, an attempt will be made to retain the more promising salient features of semi-ballistic bodies as they pertain to reliability, simplicity, flexibility and growth potential, and weight. It is recalled that the lifting vehicles presented in Phase I of this study reflected much of the Dyna-Soar technology and general vehicle shape. In a somewhat similar fashion, the semi-ballistic shape will take on certain aspects of the Mercury capsule. A notable change is the employment of aerodynamic lift during entry. This, in turn, requires rounding off the corners of the blunt face to avoid the possibility of extremely high heating rates on sharp edge. Also, rather than employ aerodynamic controls to trim the vehicle to angle of attack, the vehicle center of gravity is displaced from the center-line to provide the desired trim condition.



$X_{C.G.} = .24\ d$

$Z_C = .0515\ d$

The vehicle is controlled by rotation of the resultant force vector out of the vertical plane by rolling the vehicle around the wind axis. This rolling maneuver is accomplished by reaction jets which provide a couple about the wind axis. Other reaction jets are provided to damp undesired motions in pitch and yaw.

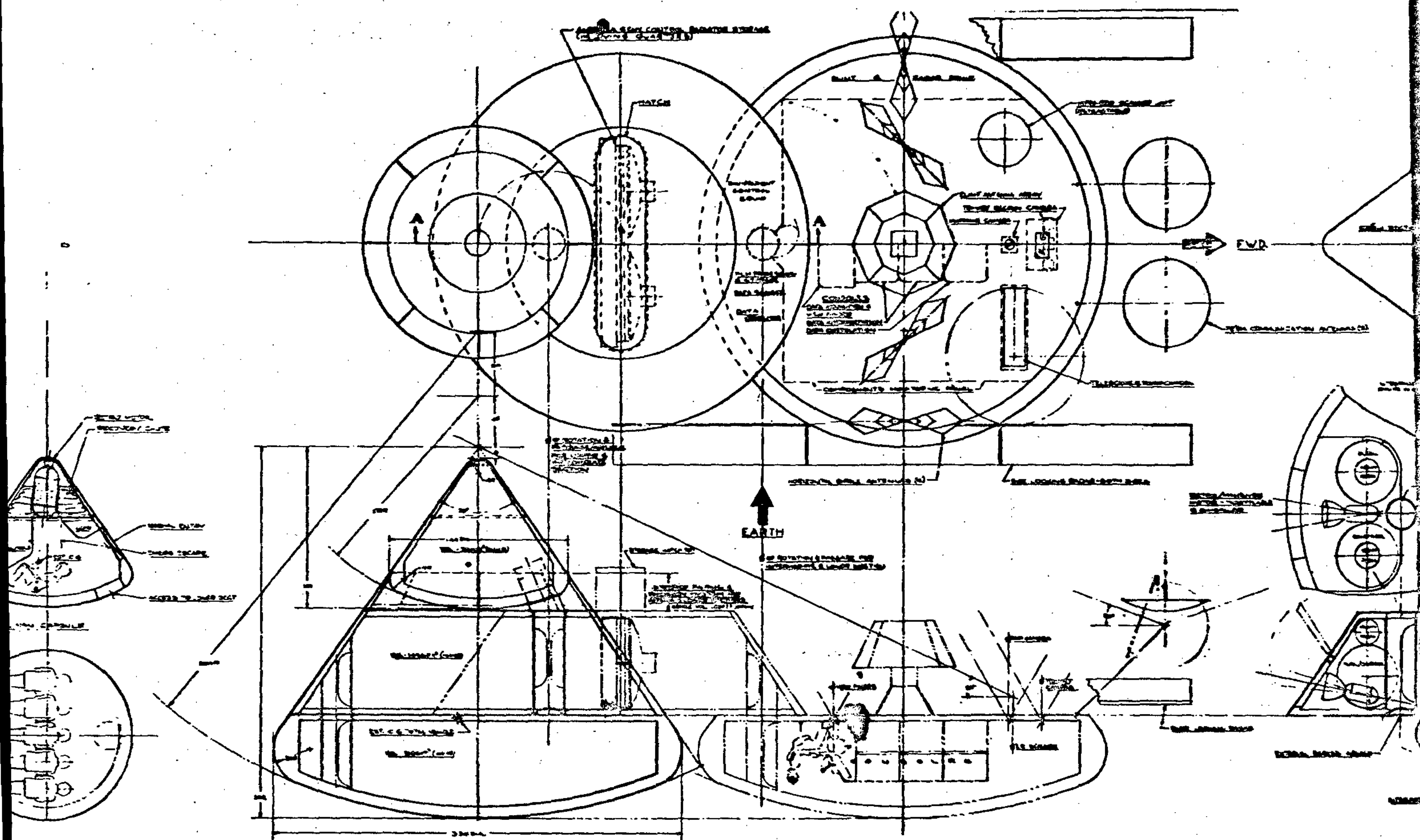
(c) Combined Arrangements

In an effort to combine the desirable features of the modular and integral arrangements in one vehicle, the following concepts evolved. Always provide the same aerodynamic shape for entry or abort, independent of mission and crew. Additional volume for payload and crew is provided by adding "rooms" to the basic vehicle. Area required to accommodate the deployed orbital mission equipment is provided by rolling out the "rooms" from the basic vehicle. For entry, the "rooms" are rolled back to the launch configuration. Figure 4 illustrates preliminary concepts of a vehicle in the launch, orbit, and entry phases of a mission. Certain features of this arrangement are as follows:

1. Mission flexibility and growth potential inherent to the modular approach is provided. Removal of one or more rooms for less ambitious missions will result in a minimum weight which must be boosted.

2. The disc-like shape of the rooms furnishes constant head room for the crew. Maximum floor space for a given wall length (circumference) is provided.

1



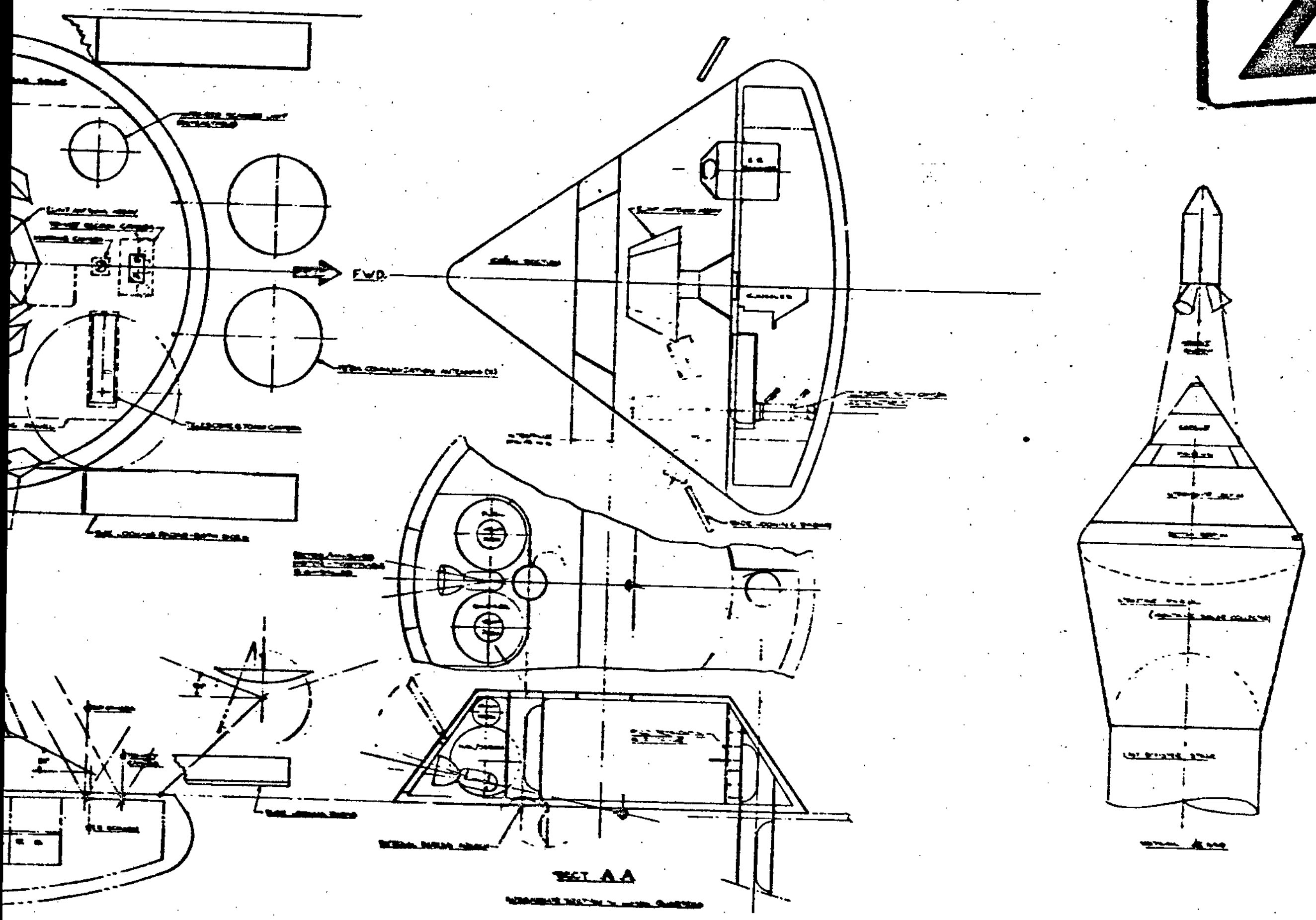


Figure 4. Preliminary Concepts of Ballistic Vehicle

3. The rate at which volume is added to the vehicle is very rapid in comparison to the increase in diameter of the blunt face.

4. The symmetrical launch stack configuration will minimize booster stability problems.

5. By hardening the blunt face of the top module, escape can be accomplished from any phase of the mission in this module. If an emergency arises in orbit the crew can enter this module, quickly disconnect from the other modules and return to earth.

6. The orbital arrangement permits ingress and egress of men and equipment through the exposed "soft" portions of the vehicle.

7. Radiators, antennas, etc., and their mounting attachments are retracted within the protection of the basic vehicle "shell" for entry and boost, thus maintaining a clean aerodynamic shape. Also, special coatings to provide desired emittance characteristics during orbit are shielded from the boost and entry environment.

8. If desired, terminal recovery (by parachute or parawing) of the individual modules may be accomplished, rather than recover the entire vehicle as an integral unit.

9. Complete system development is enhanced since the possibility of employing permanently orbiting modules with rendezvous by another vehicle is not precluded by this arrangement. Also, special experiments such as artificial gravity modules, may be accommodated.

10. The added complexity of the roll-out rooms will accrue some weight penalties. However, the elimination of wings from the vehicle may more than make up for such weight penalties.

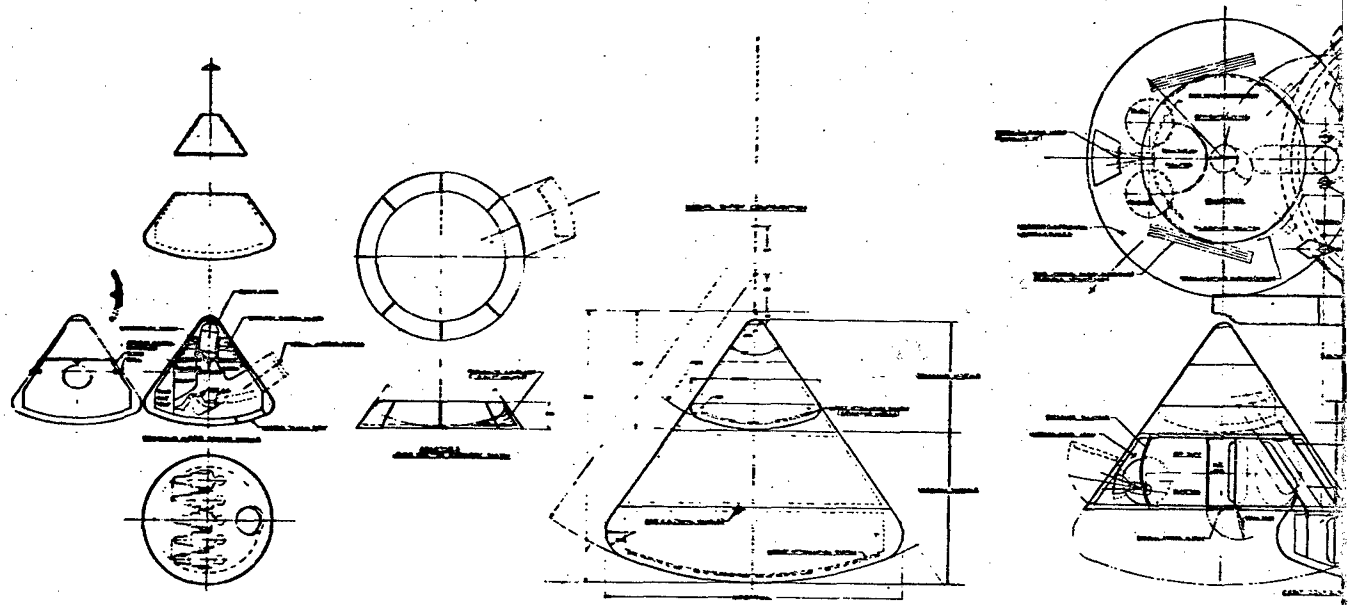
Referring again to Figure 4, a number of items are noted. One, the external antenna array does not appear to require the surface area provided by rolling out the intermediate section of the vehicle. Two, all of the available floor space and volume is really not utilized. Three, the large hatch required to permit ingress and egress of antenna and radiators is a potential leak source. Four, the manual erection of all of the antenna as well as the space radiator may be too time consuming and hazardous for the crew. These undesirable features are ameliorated as illustrated by the vehicle layout on Figure 5. A discussion of the characteristics of this base point vehicle from launch through terminal recovery follows.

Selected Vehicle

The selected base point vehicle as it is configured for various phases of a mission is shown on Figure 5. The conical shaped semi-ballistic vehicle is composed of the command or escape module which is the top portion of the cone, the mission module which is the bottom portion of the cone and contains an off-duty area and a work area, and an adaptor section which joins the two modules. This vehicle is connected to the booster through the booster fairing. On top of the entire stack is the escape tower and rocket motors. In the event of an emergency on the pad or during boost, the solid propellant escape motors would be fired, the escape module separated from the rest of the stack and accelerated away at approximately 18 g's. The total impulse is sufficient to provide the required displacement from the stack and altitude so that the escape (command) module chute recovery system will deploy and function properly.

During the boost all of the crew are located in the command module and the entire launch vehicle is controlled from this module. During this portion of the flight the men will be wearing pressurized suits and the command module will also be pressurized. The escape tower and rocket motors will normally be jettisoned during this boost portion of the flight. Once the desired orbit is attained, three of the crew will enter the off duty area of the mission module, check it for pressure integrity and activate the environmental control for this area. Next the pins which connect the work area and off-duty area are removed and the work area portion of the mission module is manually extended to position by rolling it about the connecting air-lock. This work area is now pressure checked and the environmental control activated. Next, the solar collector, which is folded within the booster fairing, is deployed by a crew member from within the vehicle and the main power system is activated. If nuclear

1



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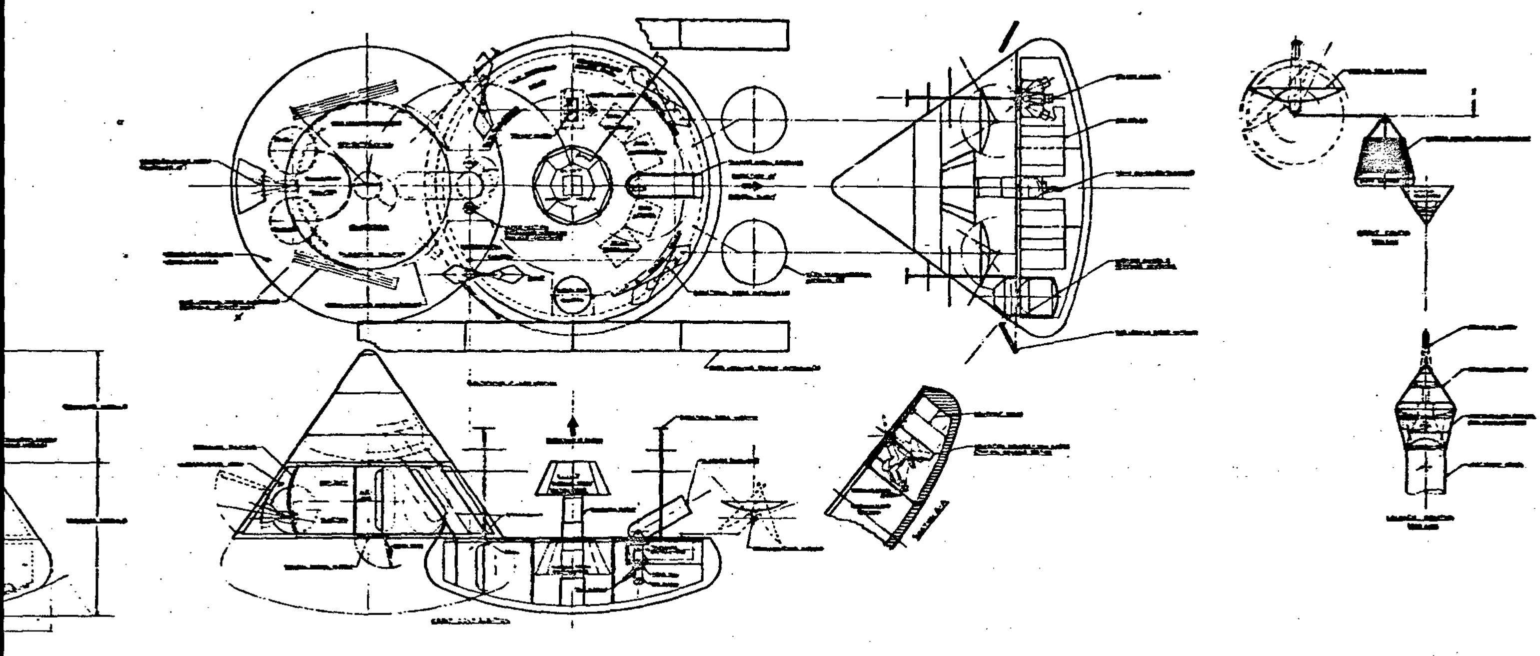

Figure 5. Base Point Ballistic Vehicle

power is employed, the reactor and shielding would likewise be stowed and deployed. It is noted that the rolling out of the work area has exposed the unpressurized portion of the off-duty area wherein various antennas are stored. A portion of the crew will now egress from the vehicle via the centrally located air-lock in the work area and erect the stored equipment. The Elint antenna array is actuated from within the work area as well as the infra-red camera package. Note that in this layout the amount of equipment to be externally deployed by the crew is kept to a reasonable minimum. The space radiator has been incorporated into the booster fairing. Also the excess area and volume previously noted have now been utilized for the Elint antenna and for non-pressurized storage areas, thus reducing potential sources of leakage. Once the surveillance equipment is activated and properly functioning, the crew will remove their pressure suits and perform the normal orbital mission duties. In the event of a serious emergency arising in orbit, the crew may quickly leave their area and enter the command module through the passageway illustrated. The command module may then be disconnected from the rest of the vehicle and de-orbited with the solid propellant rocket provided at the top of the module.

Normal maneuvering of the complete vehicle while in orbit is accomplished with hypergolic liquid propelled (5500 lbs of 50-50 N_2H_4/UDMH and N_2O_4) rocket motors composed of the gimballed engine adjacent to the off-duty area and the roll and attitude motors on the command module. Figure 6 is a sketch of the vehicle in orbit which illustrates, perhaps a little more clearly, the antenna array and general vehicle orientation. The solar collector is oriented to shade the remainder of the vehicle, including the space radiator. When the orbital mission is concluded, the solar collector and space radiator (boost fairing) are jettisoned and the rest of the equipment restowed. The work area portion of the mission module is rolled back into position and locked and the retro-grade velocity is applied by the gimballed motor adjacent to the off-duty area to initiate deorbit. Figure 7 is a sketch of the vehicle in its entry configuration.

Figure 8 illustrates an orbital configuration wherein additional radiator area is shown. These radiator surfaces, as required, may be designed to be folded up between the work area and the off-duty area.

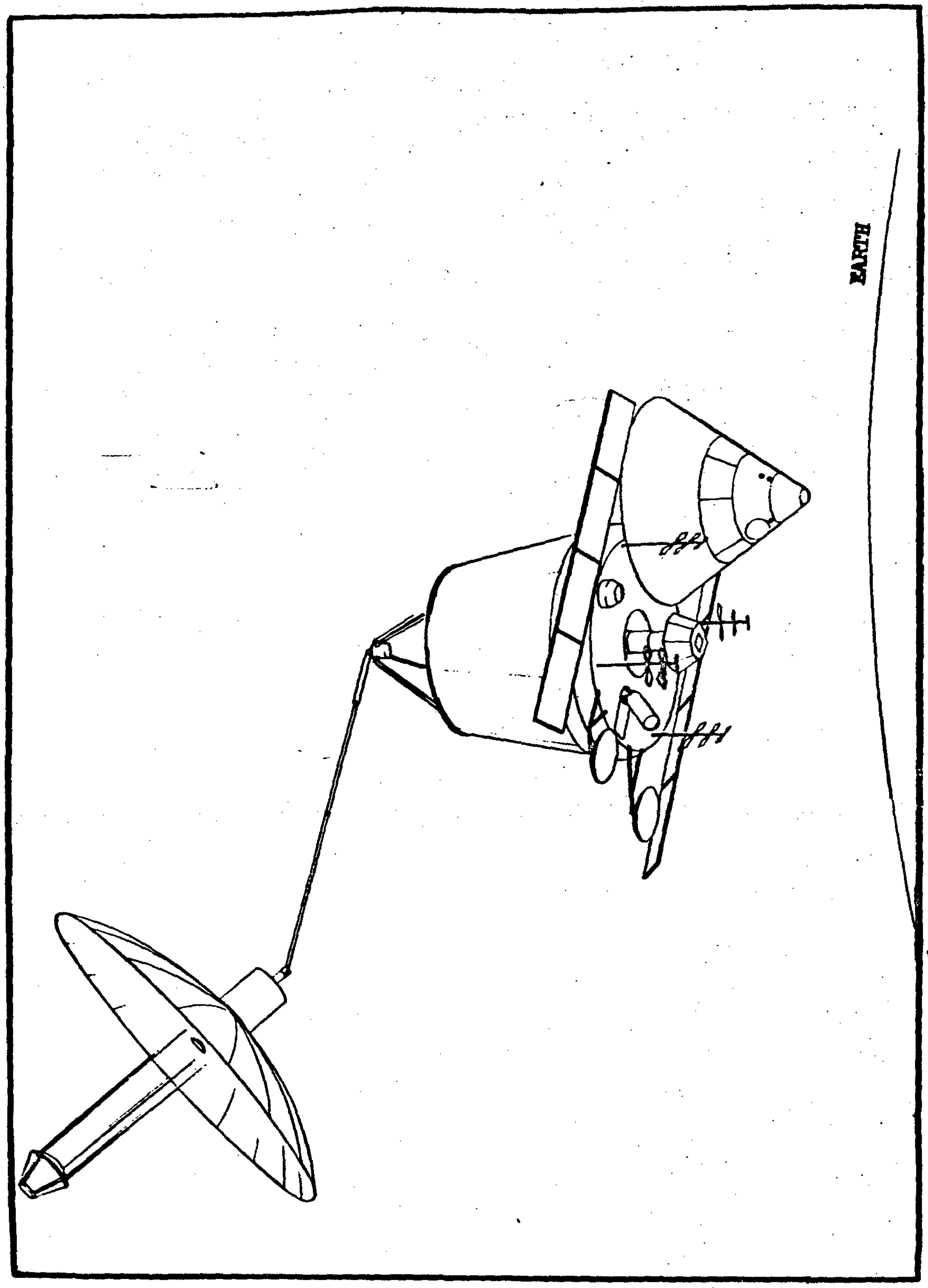

Figure 6. General Vehicle Configuration Including Antenna Array

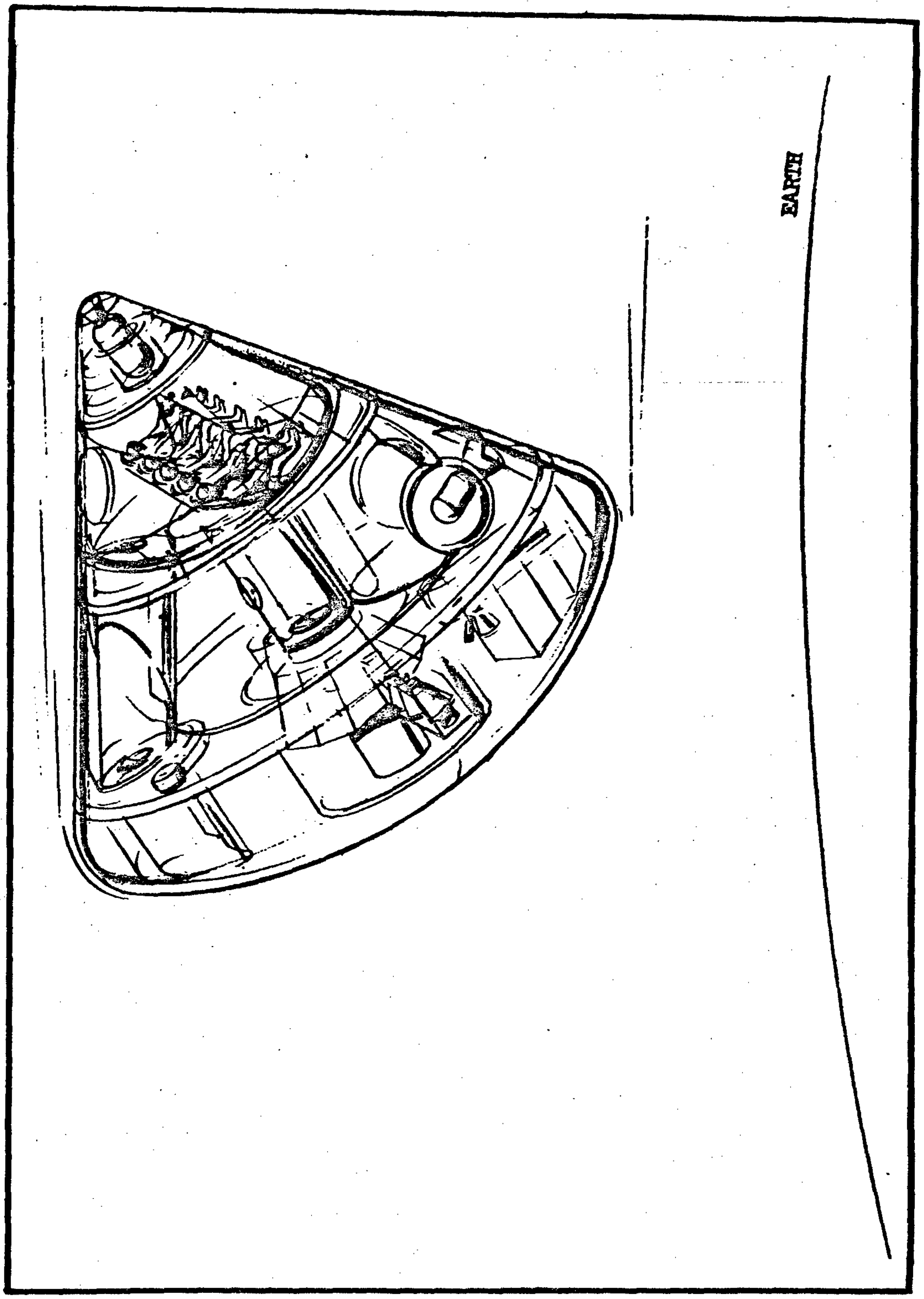

Figure 7. Ballistic Vehicle Entry Configuration

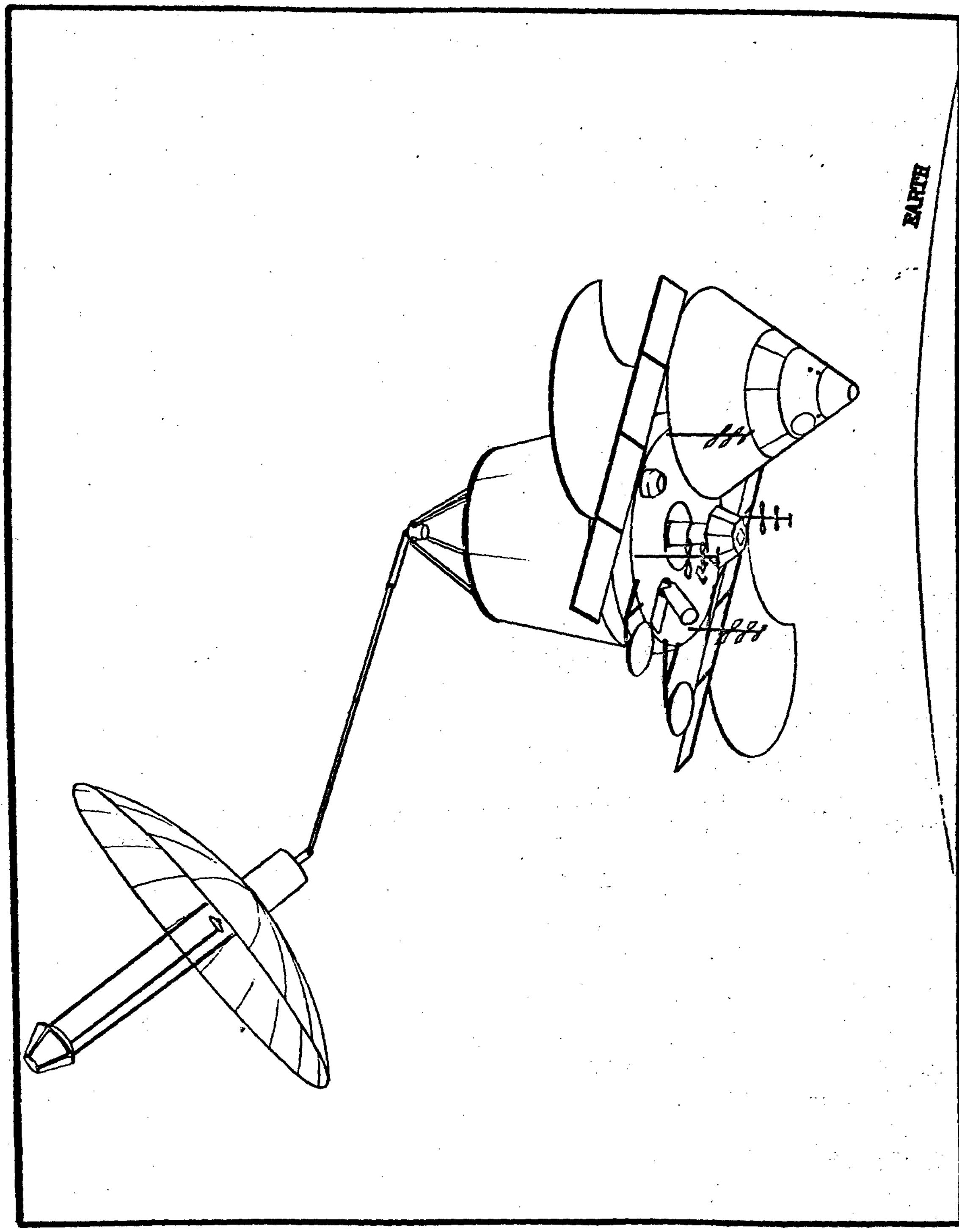

Figure 8. General Vehicle Configuration Showing Additional Radiator Area

Earth Entry

Vehicle Aerodynamics and Control

As mentioned previously, this selected vehicle is what is frequently called a lifting body. When this symmetrical body is held at an angle of attack to the relative wind, a moderate amount of lift will be produced. At an angle of attack of approximately 35 degrees the vehicle produces a lift coefficient of .51 and a drag coefficient of 1.02, hence a L/D of .5. The vehicle center of gravity is offset from the line of symmetry to produce this self-trimming capability rather than aerodynamic control surfaces. Maneuverability is provided by rolling the resultant force vector out of the vertical plane to steer right or left, or to increase or decrease the longitudinal range. However, the vehicle, by virtue of rolling about the wind axis always presents the same surfaces to the relative wind, which simplifies the heat protection system. The vehicle is rolled by reaction jets located on the command module which provide a couple about the wind axis. Undesired motions in the pitch and yaw plane are damped by additional rocket motors spaced about the command module.

Entry Environment

Figures 9 and 10 are time histories of the entry from 400,000 ft, where the entry is assumed to begin, to 100,000 ft where the terminal recovery of the vehicle will be initiated. Entry flight path angles of 5 degrees and 2 degrees are chosen as representative of a reasonable range of initial entry conditions. Also shown on these figures is the stagnation point heating rate of the vehicle and the integrated stagnation point heat load. Although the steeper entry has higher peak heating rates, the more shallow entry has a higher integrated heat load and is the more severe from the standpoint of the heat protection system. Table 3 summarizes the deorbit and entry parameters of interest.

Heat Protection System

The heat protection system selected for this vehicle employs a charring ablator. Phenolic nylon was selected for the ablation material due to its good insulating qualities. Figure 11 shows the time history during entry of the stagnation point surface temperature, backface temperature and ablation material thickness. It is desired to limit the peak backface temperature to approximately 960°R to preserve the integrity of the bond between the ablation material and the back-up structure as well as to maintain a reasonable operating temperature for the aluminum backup structure itself. It is

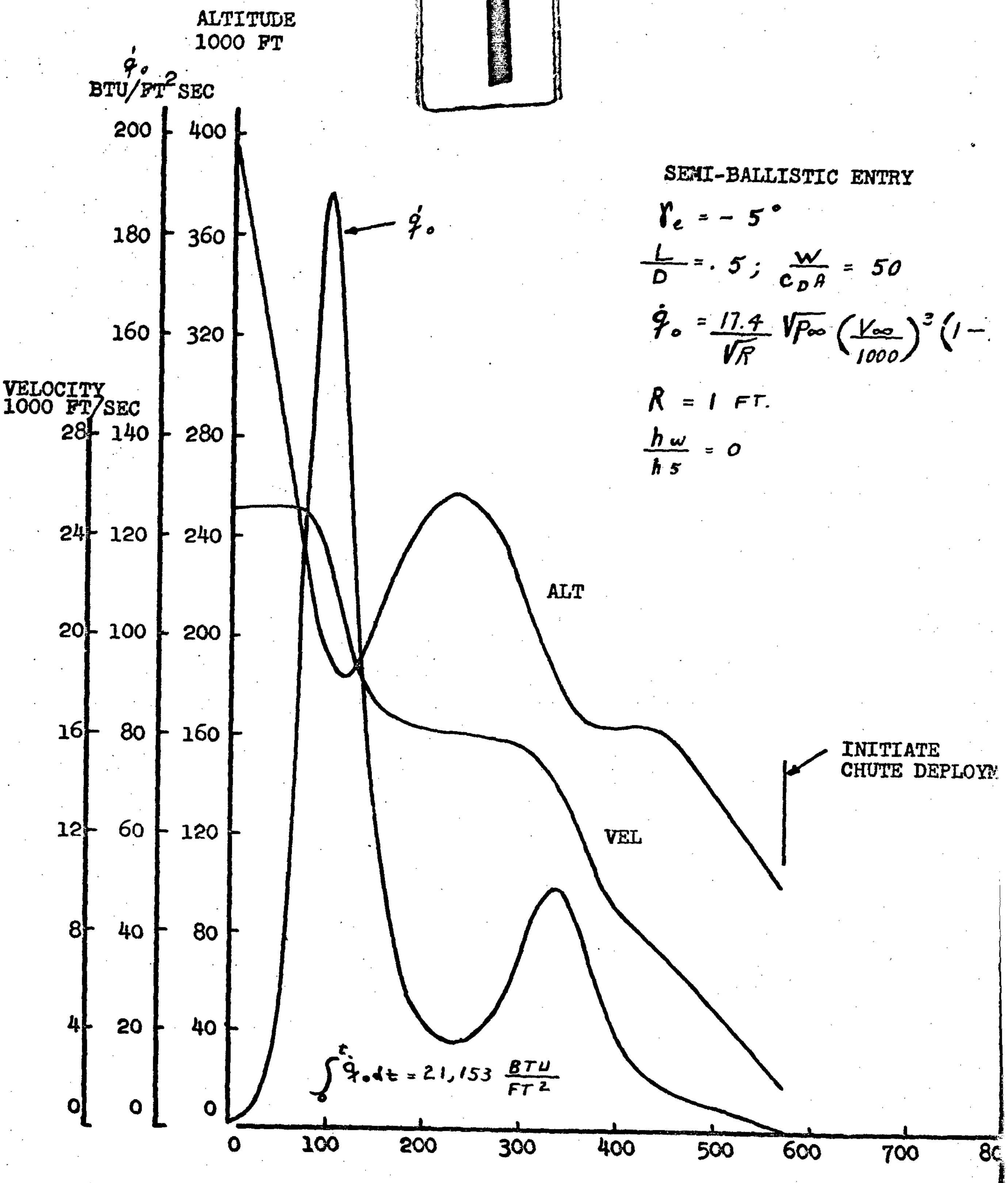

Figure 9. ENTRY TIME HISTORY - $\gamma_e = -5°$

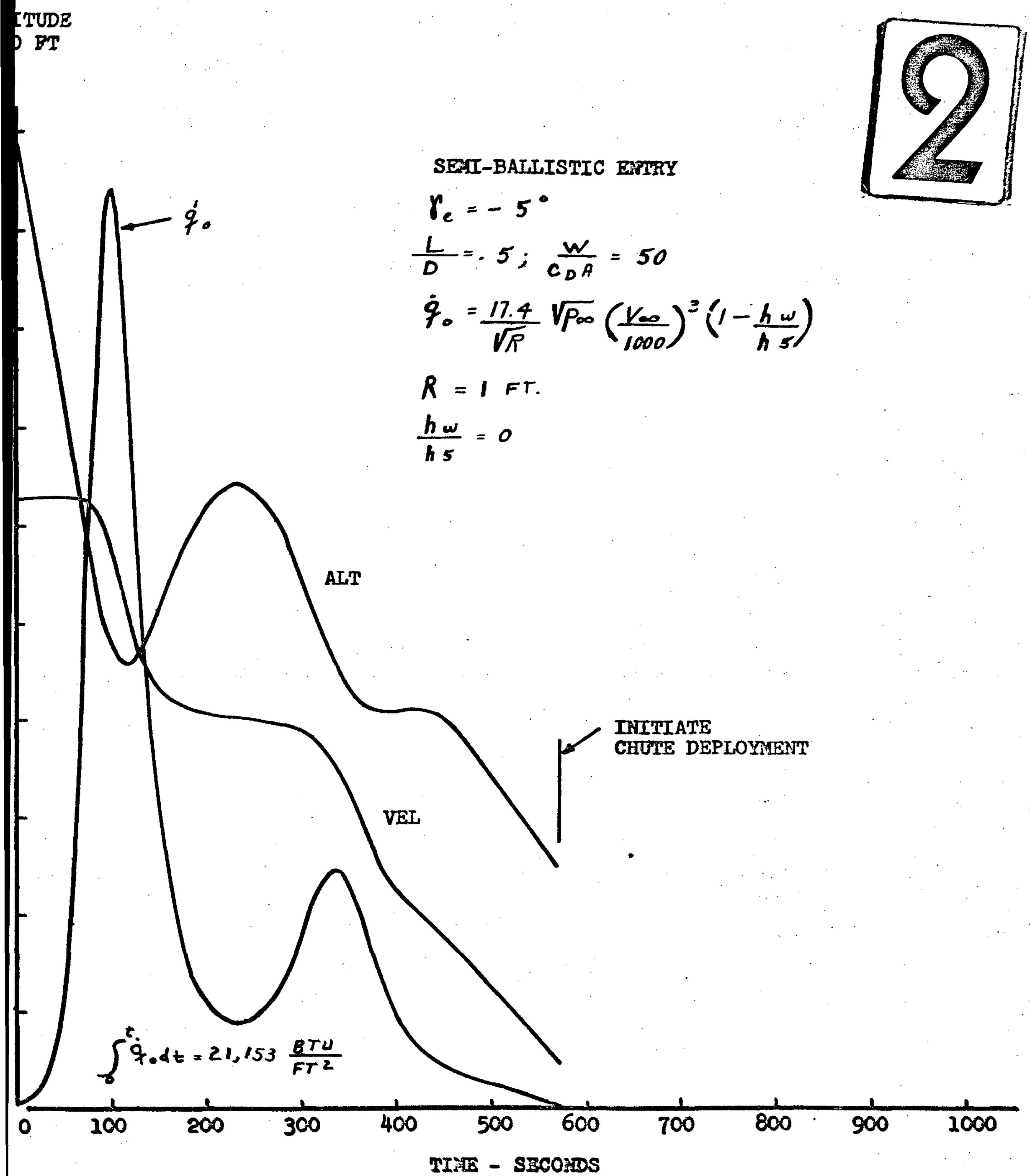

Figure 9. ENTRY TIME HISTORY - $\gamma_e = -5°$

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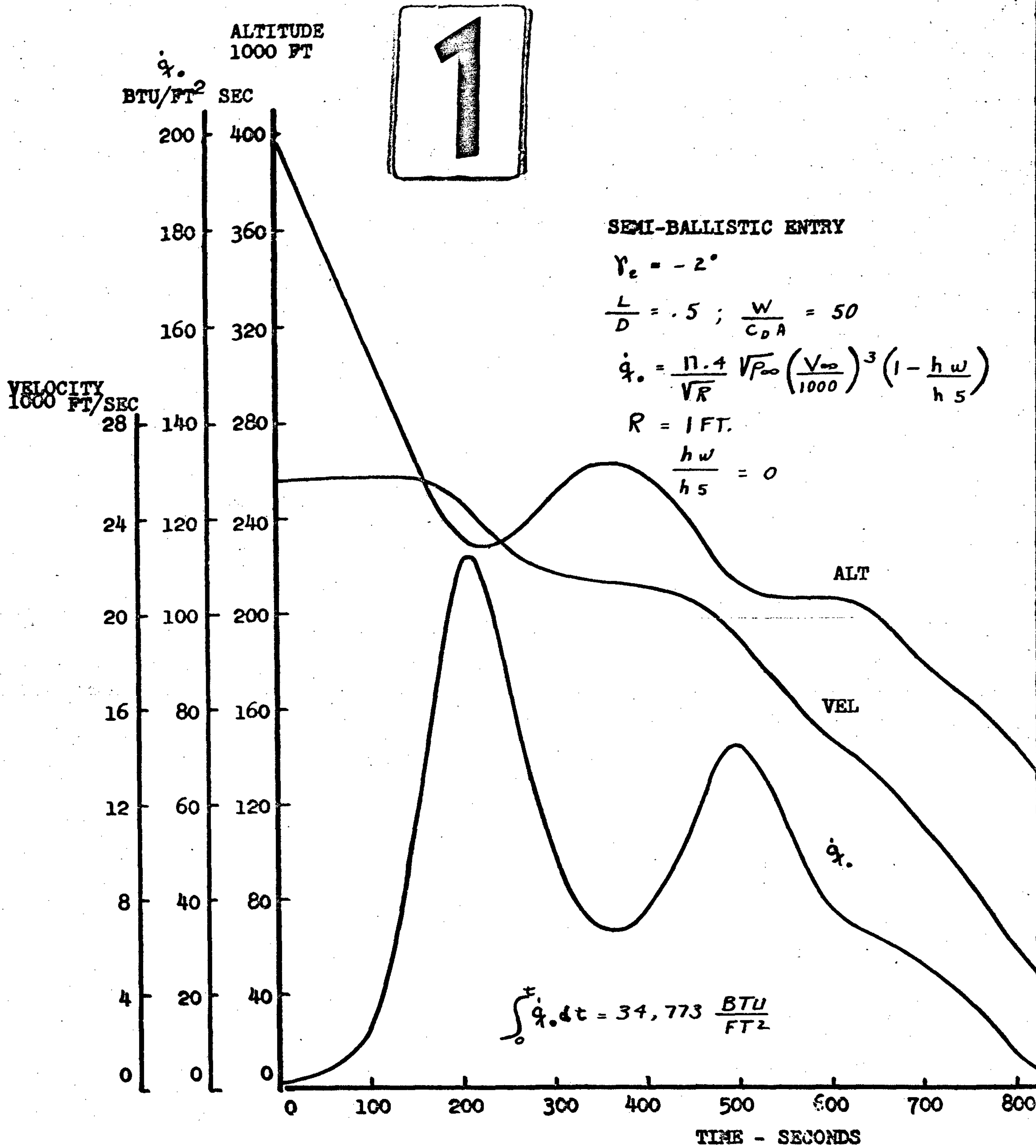

Figure 10. ENTRY TIME HISTORY - $\gamma_e = -2°$

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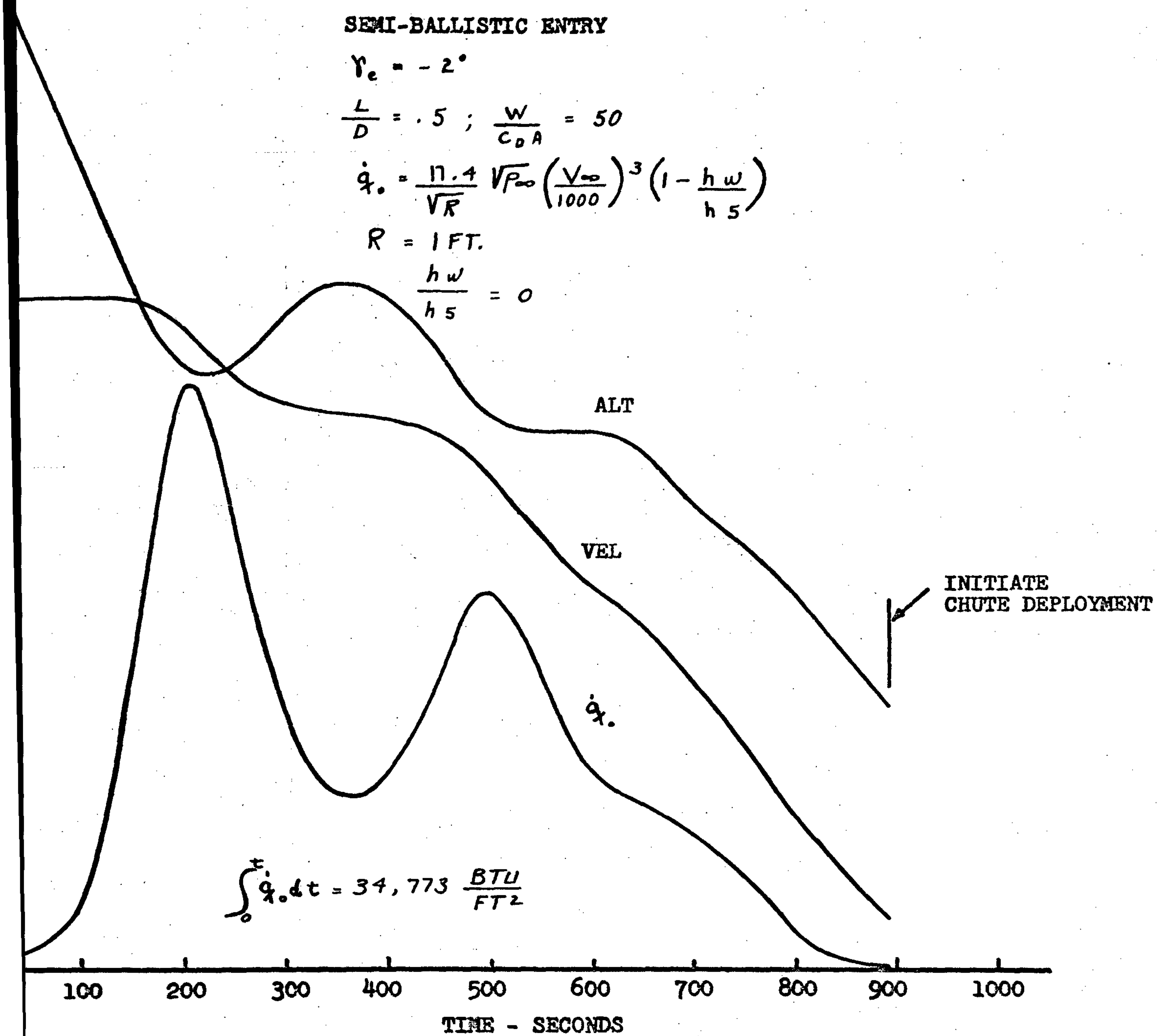


Figure 10. ENTRY TIME HISTORY - $\gamma_e = -2^\circ$

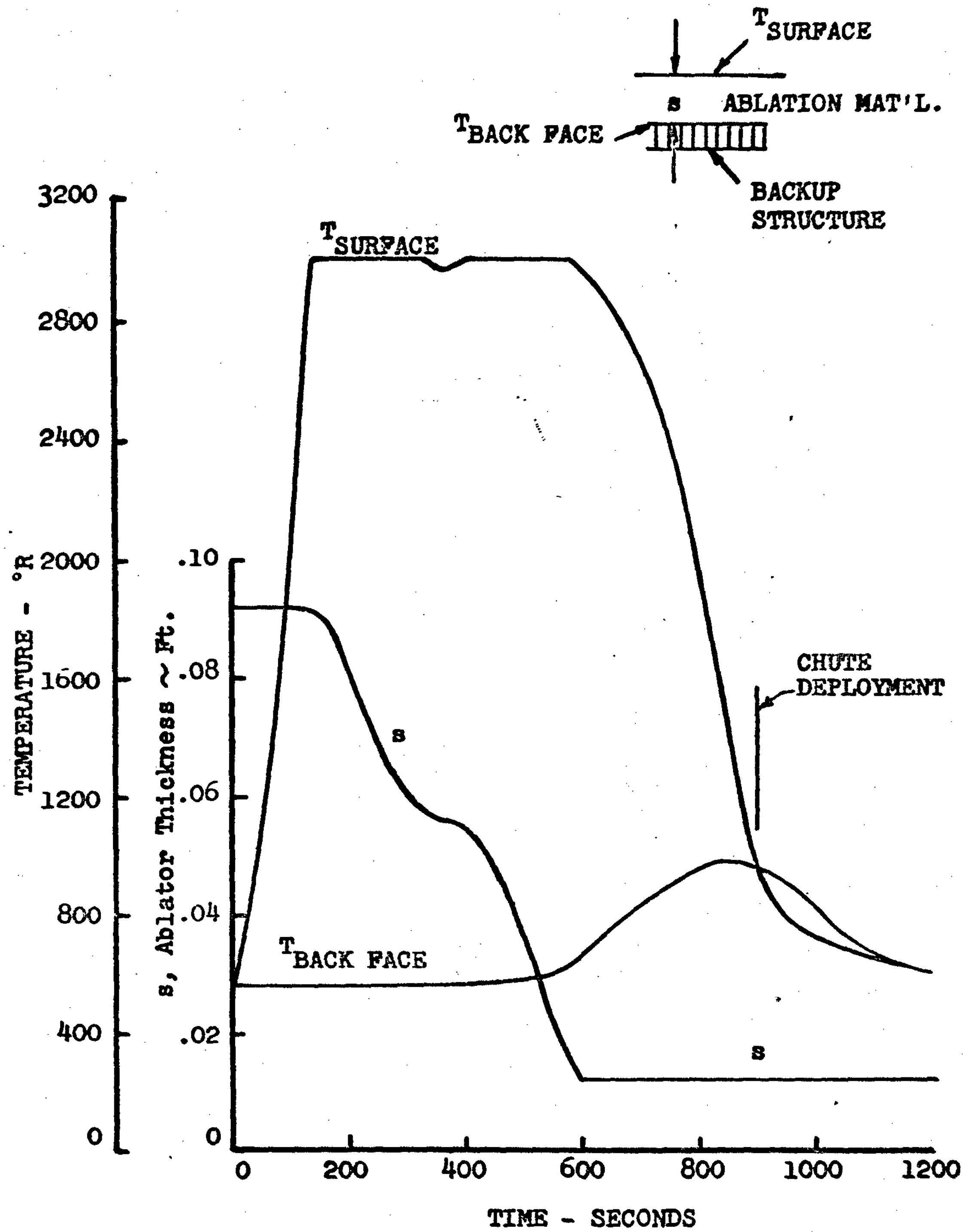


Figure 11. Stagnation Point Structure Temperatures During Entry, $\gamma_e = -2°$

TABLE 3.

DEORBIT AND ENTRY PARAMETERS

ORBIT ALT. = 300 N. MILES TO ENTRY ALT = 400,000 Ft.				ENTRY FROM ALT = 400,000 ft to 100,000 Ft.*		
Velocity Decrement Ft/sec	Elapsed Time Sec.	Distance Travelled N. Miles	Entry Flight Path Angle Deg.	Max. Resultant Load Factor G's	Elapsed Time Sec.	Distance Travelled N. Miles
550	1800	7500	-2	1.45	893	2558
1150	1100	4500	-5	5.55	570	1315

*Terminal Recovery by Parachute or Parawing Initiated at 100,000 Ft.
Descent Time is about 12-15 Minutes

seen from this figure that an ablator thickness of .0918 ft (1.1 inches) will satisfy the above requirements at the stagnation point, which is the location on the vehicle subjected to the most severe heating conditions. Integrated heat loads for other selected locations on the vehicle were computed and the ablator thickness required to limit the backface temperature to 960°F was estimated. The location of the selected points on both the mission module and the command module are illustrated on Table 4. Also tabulated are the estimated insulator thicknesses required to limit the cabin wall and impact bag temperatures to the values listed in the table. Table 5 summarizes the structural material properties and illustrates a typical section of the structural heat protection system and the primary, or pressure wall, structure.

Weight Summary

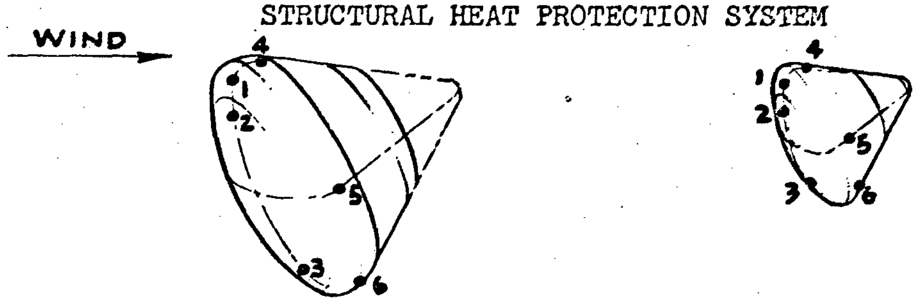
The vehicle weight summary appears on Table 6 and should be considered an estimate rather than a precise weight statement. The boosted weight of the vehicle is 43,620 lbs, and entry weight is 33,340 lbs. Figure 12 shows the approximate effect of cabin pressure on leakage rate and structure weight. This information is included as possible trade-off data for subsequent T & A considerations.

Terminal Recovery

Since the aerodynamic characteristics of this vehicle are such that a normal runway landing cannot be executed, terminal recovery of the vehicle will be accomplished by parachute. Initial drogue chute deployment is at approximately 100,000 ft. The command module which contains the crew will separate from the rest of the vehicle first and recover separately. The mission module, whose chutes have been stowed in the adaptor section between the two modules, will similarly be recovered. Prior to the impact with the ground or water, the heat shield structure on the faces of the command and mission modules will be extended (dropped down) thereby inflating the impact bags with ambient air pressure. On impact, these bags which are equipped with blowout panels, will cushion the shock of landing. As an alternate consideration, the command module may be equipped with a parawing rather than a chute to provide additional maneuverability during the terminal recovery. In this case, it may be required to supply ejection seats for the crew for off-the-pad escape.

TABLE 4.

STRUCTURAL HEAT PROTECTION SYSTEM



MISSION MODULE　　　COMMAND AND ESCAPE MODULE

Point No.	Mission Module: Ablator[1] Thickness Inches	Mission Module: Insulator Thickness Inches	Command and Escape Module: Ablator Thickness Inches	Command and Escape Module: Insulator Thickness Inches
1	1.10	0.5[2]	1.36	0.5[2]
2	0.62	0.5[2]	0.77	0.5[2]
3	0.47	0.5[2]	0.59	0.5[2]
4	0.53	1.0[3]	0.66	2.25[4]
5	0.20	1.0[3]	0.25	2.25[4]
6	0.125	1.0[3]	0.125	2.25[4]

(1) Required to limit ablator backface temperature to 500°F

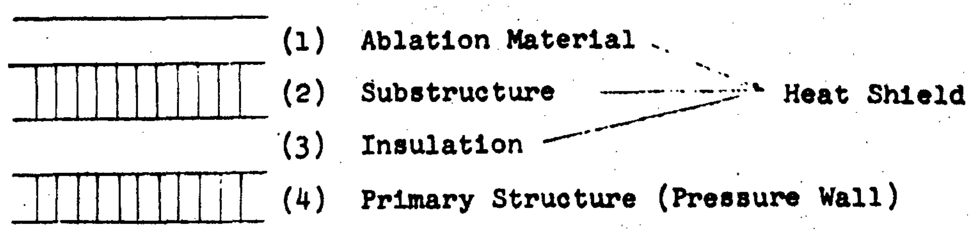
(2) Required to limit impact bag temperature to 300°F

(3) Required to limit unoccupied cabin wall temperature to 200°F

(4) Required to limit occupied cabin wall temperature to 130°F

TABLE 5.

STRUCTURAL MATERIAL PROPERTIES



Item No.	Material	Specific Weight (Note Dimensions)	Specific Heat BTU/lb °F	Thermal Conduct. BTU/Ft.HR.°F	Emittance	h_{eff}
1	Phenolic Nylon	76 lbs/ft^3	0.45	0.14	0.8	(1)
2	Aluminum Honeycomb	1.5 lbs/ft^2	(2)	(2)	-	-
3	Micro-Fibres Felt	4 lbs/ft^3	0.25	0.05	-	-
4	Aluminum Honeycomb	2* lbs/ft^2	(2)	(2)	-	-

*Estimated at 10 PSI internal pressure

(1) Effective heat of ablation was taken to be

$h_{eff} = \frac{h_s}{2}$ where h_s is the stagnation enthalpy

(2) For preliminary analyses, the presence of the aluminum may be ignored

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TABLE 6.

BALLISTIC VEHICLE WEIGHT SUMMARY

DESCRIPTION	ESCAPE SYSTEM DETAIL	ESCAPE SYSTEM GROUP	ESCAPE SYSTEM CONSUMED	COMMAND MODULE DETAIL	COMMAND MODULE GROUP	COMMAND MODULE CONSUMED	MISSION MODULE DETAIL	MISSION MODULE GROUP	MISSION MODULE CONSUMED
STRUCTURE		595			2800			10625	
Pressure Walls				595			3000		
Substructure				645			2330		
Escape Tower	595			85					
Windows & hatches				150					
Internal structure				75			500		
Airlocks							300		
Insulation				200			500		
Ablation Material				1050			3995		
LANDING SYSTEM								1500	
Chutes					525		1400		
Parawing				475					
Impact bags, etc.				50			100		
ROCKET MOTORS		3250			630			500	
Launch pad abort (solid)	3250		2750						
Orbit abort (solid)				580		500			
Orbit maneuver; de-orbit							500		
Entry roll control, pitch and yaw damper				50					
Subtotal - Forward		3845	2750		3955	500		12625	

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TABLE 6 (CONT).

BALLISTIC VEHICLE WEIGHT SUMMARY

DESCRIPTION	ESCAPE SYSTEM DETAIL	ESCAPE SYSTEM GROUP	ESCAPE SYSTEM CONSUMED	COMMAND MODULE DETAIL	COMMAND MODULE GROUP	COMMAND MODULE CONSUMED	MISSION MODULE DETAIL	MISSION MODULE GROUP	MISSION MODULE CONSUMED
Subtotals Brought Forward		3845	2750		3955	500		12625	
PROPELLANTS					245			5520	
Orbit maneuver & deorbit				100		100	5400		5400
Entry Control				100		100			
Trapped liquids & gases				5			20		
Tanks and plumbing				40			100		
ELECTRICAL POWER SYSTEMS					350			1500	
Orbit-sollar collector							1500		
Launch & entry-batteries				350					
ELECTRONICS & SURVEILLANCE EQUIP.					140			6195	
Side-looking radar							650		
ELINT							1500		
Infra-red							250		
Cameras							1000		
Processor							200		
Film							700		
Solutions							700		
Data Processing							1000		
Communications Transmitter							150		
Communications Receiver							30		
Inertial platform				40					
Guidance computer				100					
Horizon seeker							15		
Subtotal - Forward		3845	2750		4680	700		25840	5400

TABLE 6 (CONT).

BALLISTIC VEHICLE WEIGHT SUMMARY

DESCRIPTION	ESCAPE SYSTEM DETAIL	ESCAPE SYSTEM GROUP	ESCAPE SYSTEM CONSUMED	COMMAND MODULE DETAIL	COMMAND MODULE GROUP	COMMAND MODULE CONSUMED	MISSION MODULE DETAIL	MISSION MODULE GROUP	MISSION MODULE CONSUMED
Subtotals Brought Forward		3845	2750		4680	700		25840	5400
MOUNTING AND DISPLAY PANELS					300			900	
CREW AND CREW SUPPORT					1890			760	
Crew (5)				970					
Suits and accessories				115					
Back packs				150					
Seats				380					
Restraint				15					
Seat supports				110					
Food and containers							450		
Water and containers							275		
Relief provisions							35		
Survival kit				140					
Bio-medical instruments				10					
THERMAL CONTROL SYSTEM					85			1040	
Heat Exchangers, blowers, etc.				60			240		
Space Radiators							800		
Expendable heat sink				25		25			
ATMOSPHERIC CONTROL SYSTEM					110			1170	
Atmosphere Supply (H_2O_2)				30		10	570		200
Contaminate removal				80			600		
Subtotal - Forward		3845	2750		7065	735		29710	5600

TABLE 6 (CONT).

BALLISTIC VEHICLE WEIGHT SUMMARY

DESCRIPTION	ESCAPE SYSTEM DETAIL	ESCAPE SYSTEM GROUP	ESCAPE SYSTEM CONS.	COMMAND MODULE DETAIL	COMMAND MODULE GROUP	COMMAND MODULE CONS.	MISSION MODULE DETAIL	MISSION MODULE GROUP	MISSION MODULE CONSUMED
Subtotals Brought Forward		3845	2750		7065	735		29710	5600
OTHER		100			100				
Adapter to launch stack								1800	
Miscellaneous	100						800	1000	
TOTAL		3945	2750		7165	735		32510	5600

(1) Space radiator and adapter form an integral structural unit.

(2) Leakage rate of 5 lbs/day

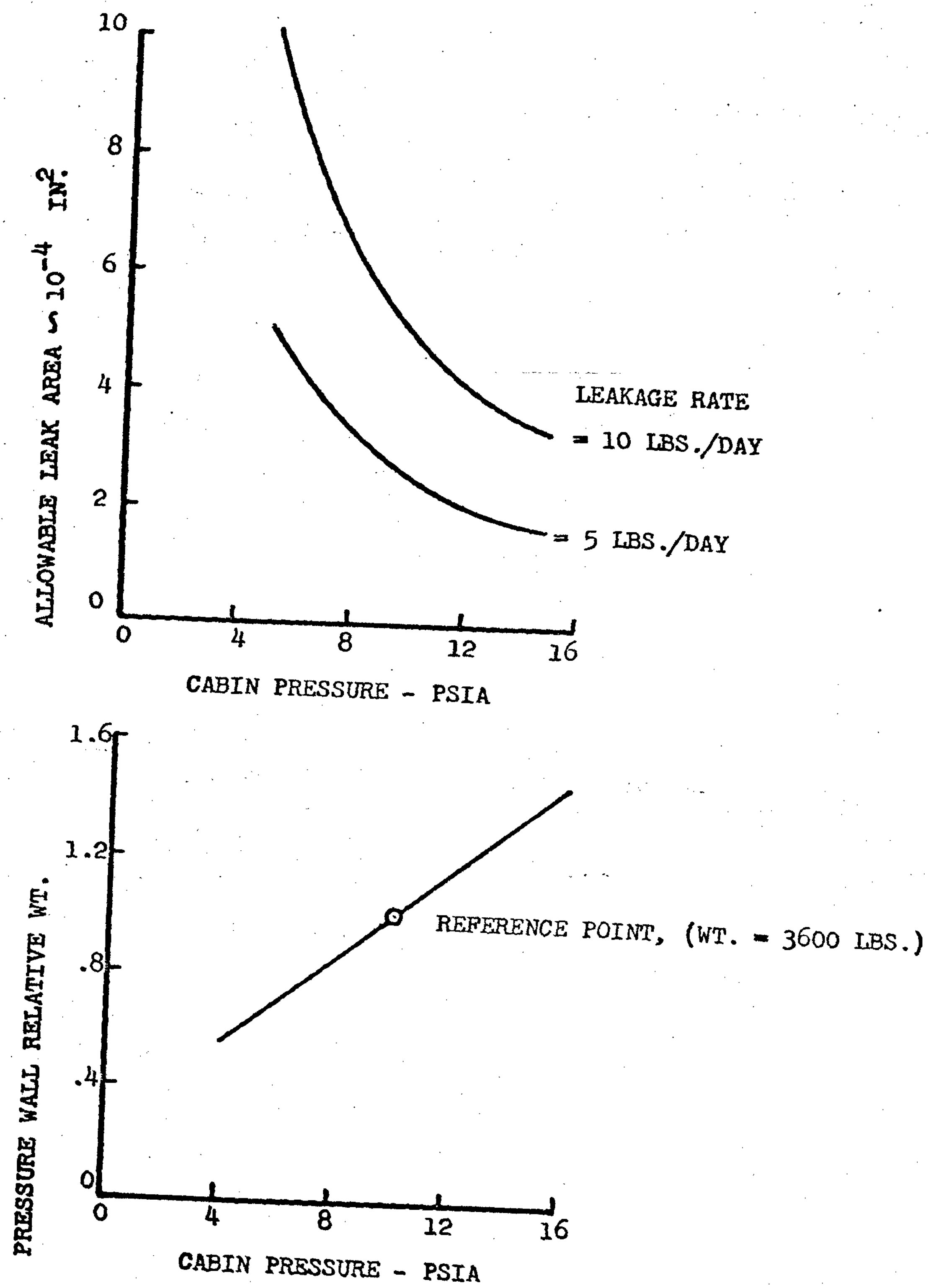

Figure 12. APPROXIMATE EFFECT ON CABIN PRESSURE ON LEAKAGE AND STRUCTURAL WEIGHT

Mission Power Demand

Table 7 indicates the estimated mission power demand for all phases of the mission including a post landing period of 72 hours. The relatively low power demand during boost and entry is due to the simple reaction motor control system employed rather than the relatively high power demand hydraulic system which may be employed if aerodynamic control surfaces were utilized.

Other Vehicles

The design of the base point vehicle is such that the vehicle may be easily modified to accommodate missions less severe than the one described above. Table 8 indicates the effect of mission and crew number on the vehicle pressurized area and volume. The first mission and crew listed correspond to the base point vehicle described herein. The second mission and crew requirement is satisfied by the base point vehicle with its work area removed. Its off-duty area now becomes the work area for this mission. The smaller crew may utilize the command module itself for off-duty area. The third mission and crew listed may be accomplished with the command module alone.

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TABLE 7.

MISSION POWER DEMAND
FOR BALLISTIC VEHICLE

MISSION PHASE	KW	HOURS	KW HOURS
Pre-launch	1.5	2 (1)	3.0
Boost	1.5 (2)	0.2	0.3
Deploy Mission Equipment	1.5	2	3.0
Orbit	22	1008	22167.0
Stow Mission Equipment	1.5	2	3.0
Orbit to Start of Entry	1.5	0.5	.75
Earth Entry	1.9	0.25	.475
Terminal Recovery	1.2	0.25	.3
Post Landing	.1	72	7.2

(1) Ground service power used for excess times

(2) Primary power for boost control is assumed to be provided by the booster itself.

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TABLE 8.

EFFECT OF MISSION AND CREW NUMBER ON VEHICLE PRESSURIZED AREA AND VOLUME

MISSION AND CREW	VEHICLE COMPONENT	PRESSURIZED AREA - Ft^2	PRESSURIZED VOLUME-Ft^3
6 Week Surveillance Mission, 5 Man Crew	Command Module	300	330
	Living Area	634	1025
	Work Area	853	1732
	TOTAL	1787	3087
6 Week Surveillance Mission, 2 Man Crew	Command and Living Module	300	330
	Work Area	777	775
	TOTAL	1077	1105
36 Hour, Limited Mission Equipment, 3 Man Crew	Command, Living, and Work Area in One Module	300	330

LENTICULAR REENTRY VEHICLE

The overall weapon system concept results in a requirement for three basic orbiting components. First, there is a requirement for a manned bombardment vehicle which houses the basic control function in space. Secondly, a weapon cluster is required. This is an unmanned weapon carrier which combines and integrates several weapons into a common orbiting package to facilitate handling and servicing. The third requirement is the weapon itself (Reference 6).

Manned Bombardment Vehicle

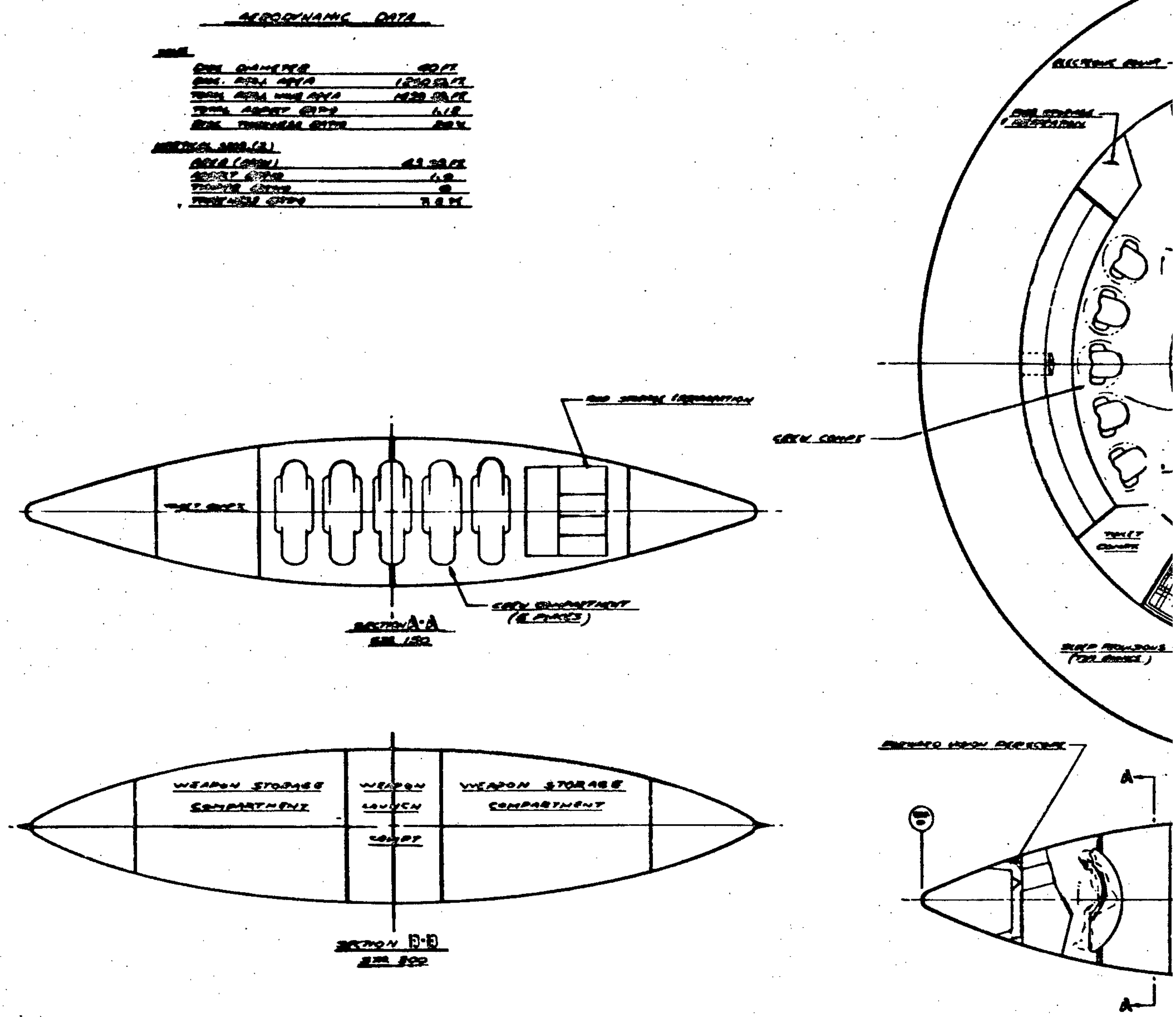
The bombardment and control vehicle is the primary element of the system, since it actually contains weapons within itself as well as exercises control of the remainder of the weapons clustered near it. This vehicle, housing the human element and the basic operational equipment, is the only portion of the system which is required to effect repeated reentries through the earth's atmosphere. Thus, the design criteria for this element of the system is much more severe than those criteria imposed on the other vehicles. The lenticular control vehicle, then, becomes the system of primary concern in describing conceptual designs for thermal and atmospheric control subsystem studies.

The structural criteria presented here are primarily based upon the temperature and loading requirements associated with the reentry phase of the mission and upon the injection boost loading. Secondary requirements include adequate insulation characteristics to assist in the control of the internal environments, and special situations, such as meteoroid encounter and crew escape provisions.

General Arrangement

The manned bomber is a lenticular shaped reentry vehicle as shown in Figure 13. The disc-shaped configuration was chosen for its greater usable volume available for weapon storage and crew accommodations and for other advantages. The volumetric efficiency of the disc with respect to a cylindrical body is shown graphically in Figure 14. It has a basic diameter of 40 feet and a gross launch weight of about 45,000 pounds. The vehicle functions as a manned orbital bombing system with an internal armament load of four winged reentry weapons and also acts as an orbital control and maintenance center for additional unmanned weapon clusters.

1


AERODYNAMIC DATA
CREW COMPARTMENT
SECTION A-A
WEAPON STORAGE COMPARTMENT
WEAPON LAUNCH COMPT
WEAPON STORAGE COMPARTMENT
SECTION B-B
CREW COMPT

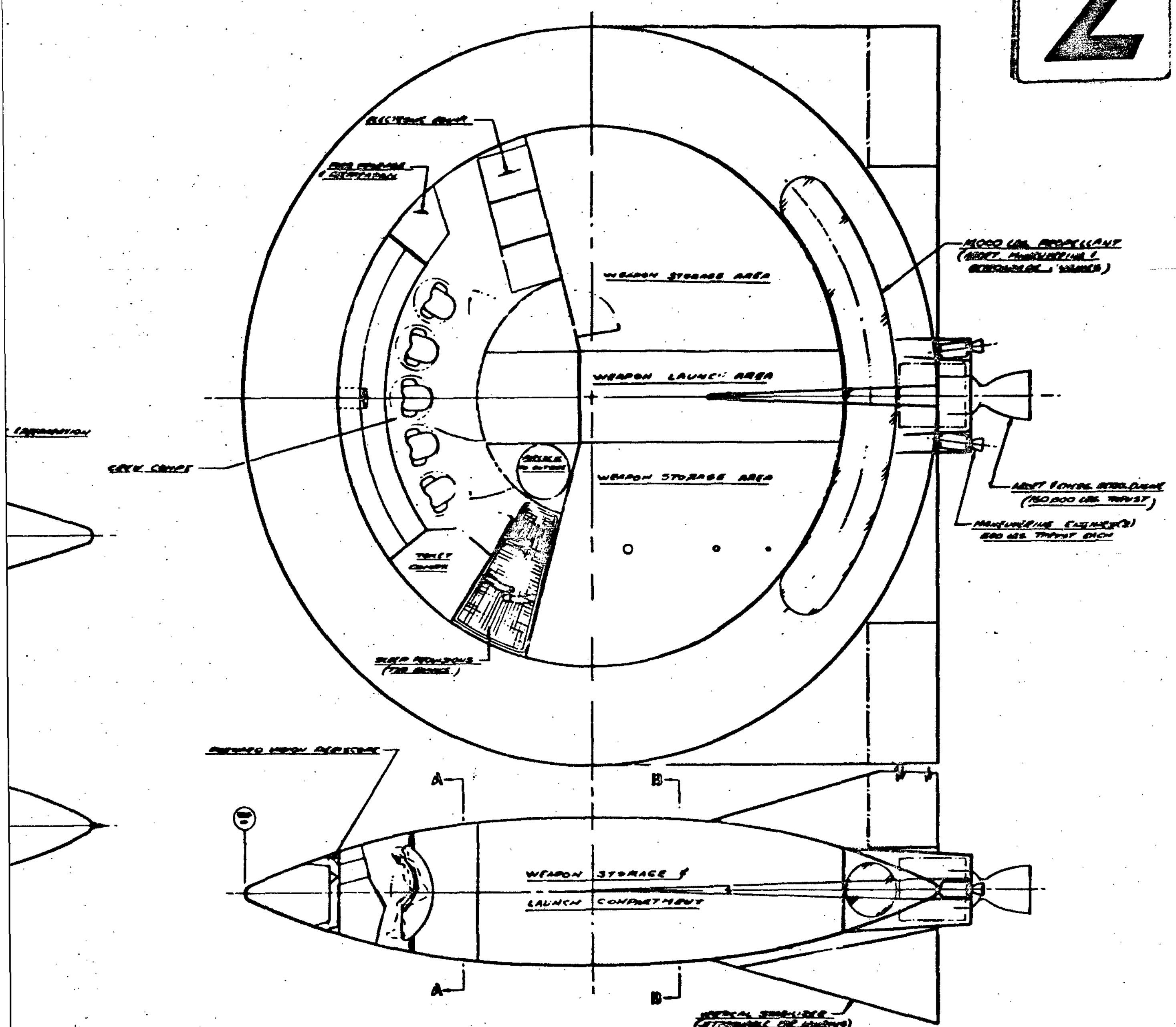


Figure 13 Disc Shape Design Study

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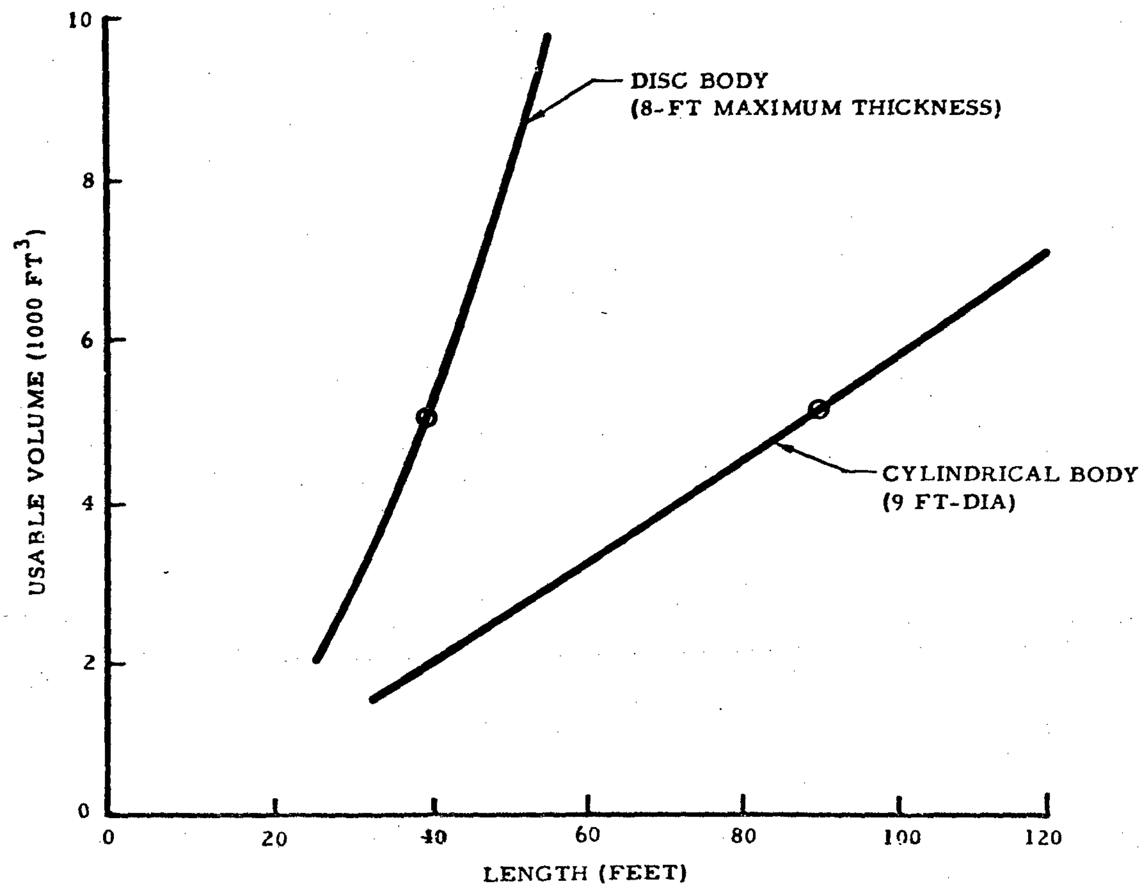

Figure 14. Volumetric Efficiency

The operational mission design is 6 weeks duration at a nominal orbital altitude of 300 nautical miles, with a crew of four men. The vehicle incorporates a jettisonable crew compartment capsule for emergency crew escape and a space shuttle service vehicle for inspection and maintenance of the weapon clusters.

External Configuration

The external configuration of the manned bomber is shown in Figure 15. The total projected planform area is 1548 square feet which results in a normal reentry and landing wing loading of approximately 23 pounds per square foot. The projected planform area noted includes the horizontal area between the disc and the straight trailing edge, which provides the flap area and also includes the movable horizontal stabilizer surface located outboard of the vertical stabilizer surfaces. The hinged trailing edge surfaces inboard of the verticals function as elevators during subsonic flight and landing. The all movable horizontal surfaces located outboard of the verticals can be actuated differentially to provide roll control as well as simultaneously to provide pitch control, and are effective throughout the entire flight regime. The landing gear consists of retractable skids mounted on the lower fuselage surface.

With the gross launch weight of the manned bombardment vehicle at 45,000 pounds, the useful load is 27,958 pounds, including 8056 pounds for four winged weapons. A weight summary of the manned vehicle including a weight breakdown of the major components is presented in Table 9.

Internal Arrangement

The internal arrangement of the vehicle is shown in Figure 16. The vehicle is designed with four primary internal compartments; the living quarters, work area, armament bay, and crew escape capsule. Each compartment can be individually isolated from the other three, should a puncture or leak make one temporarily unusable. Repairs could then be made to the damaged compartment while the crew used the remaining compartments for living and working quarters.

Crew Escape Capsule

The crew capsule functions as the vehicle control center during normal operation and as the crew escape capsule for an emergency abort of the mission. The escape capsule is actually a breakaway nose section of the vehicle, approximately 17 feet long and 6 feet wide. The general arrangement of the escape capsule is shown in Figure 17, and a weight summary is presented

1

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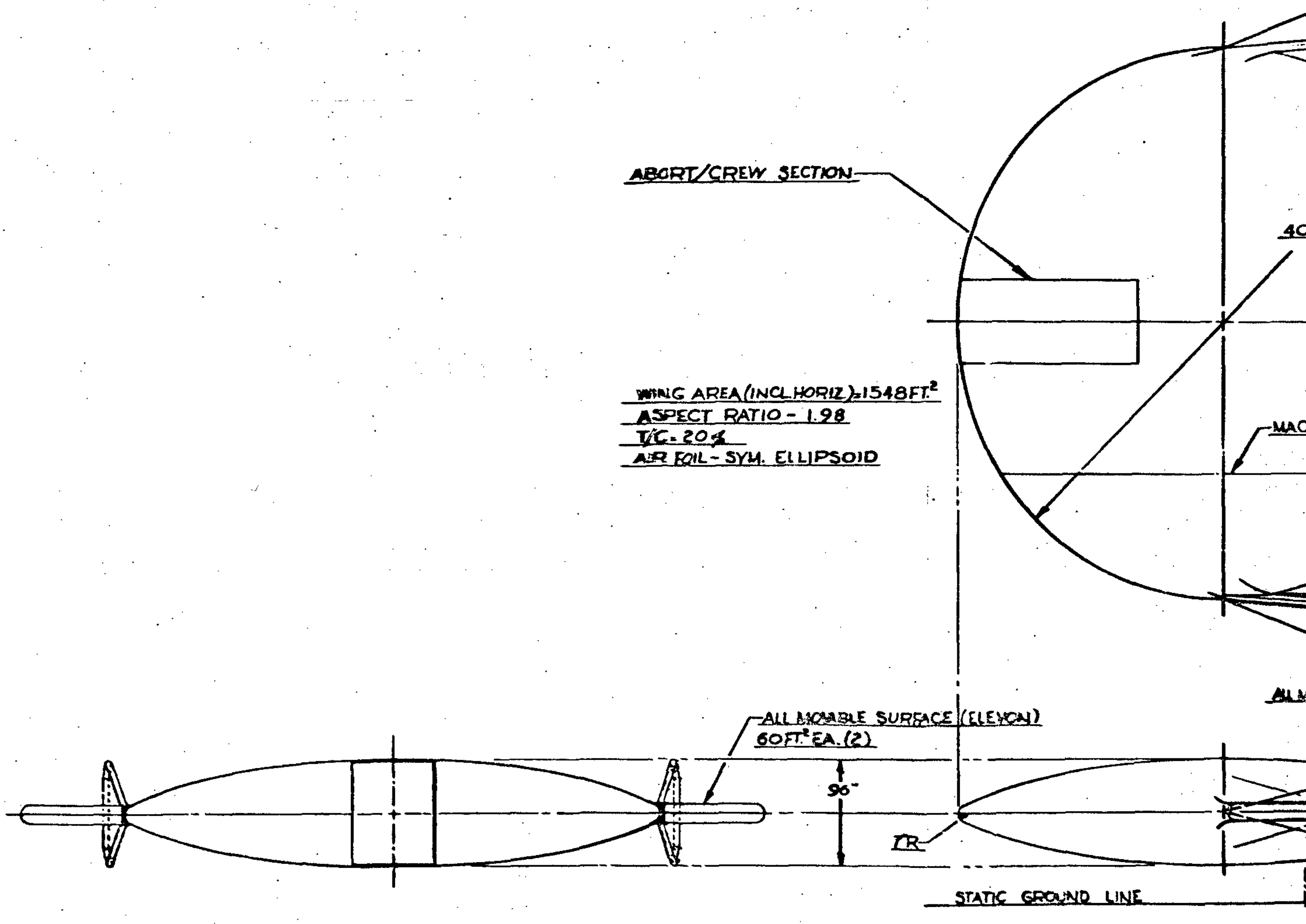


Figure1

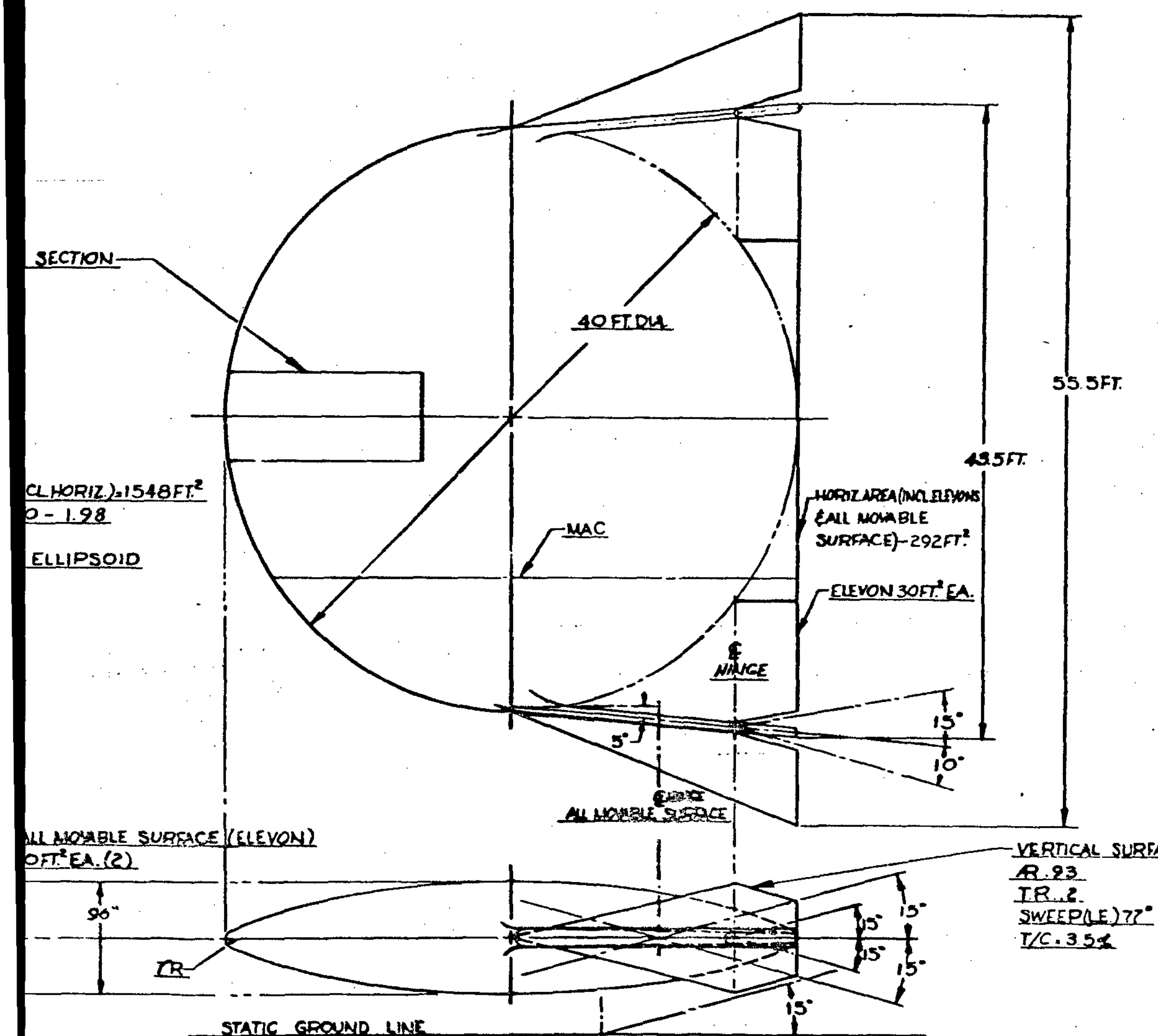
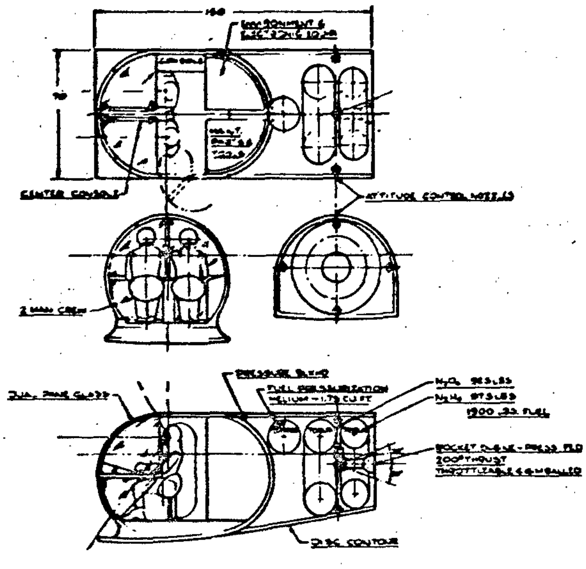
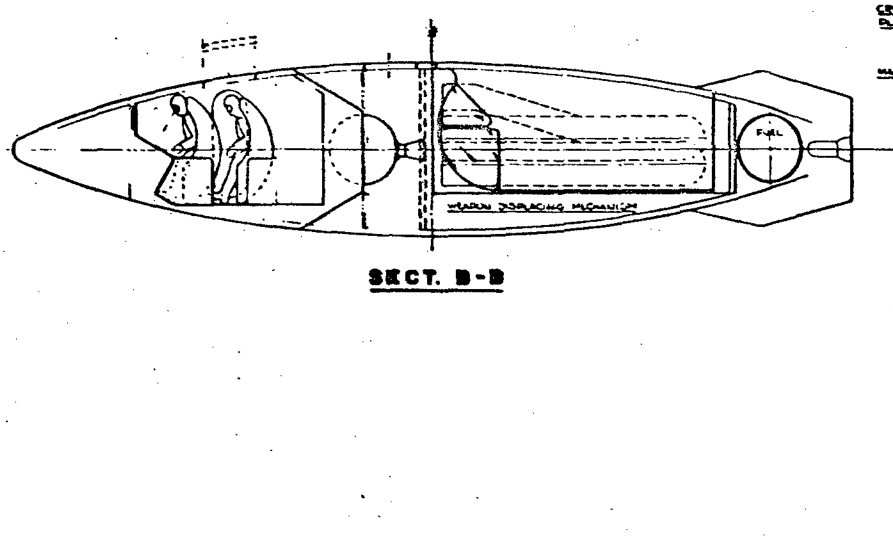


Figure 15. Manned Bombardment Vehicle General Arrangement

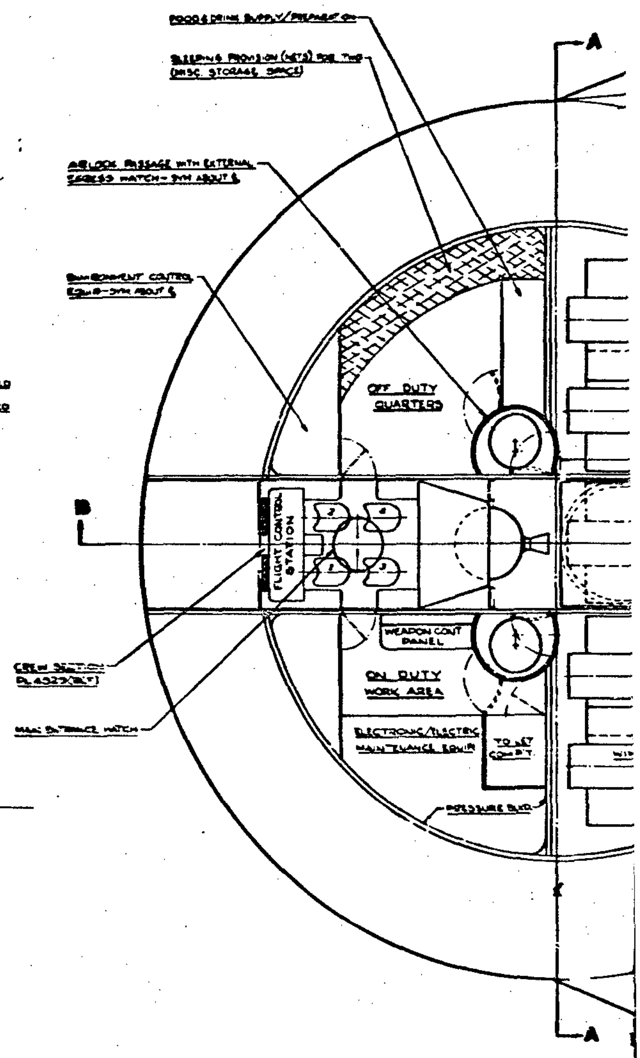
1



SHUTTLE



SECT. B-B



2

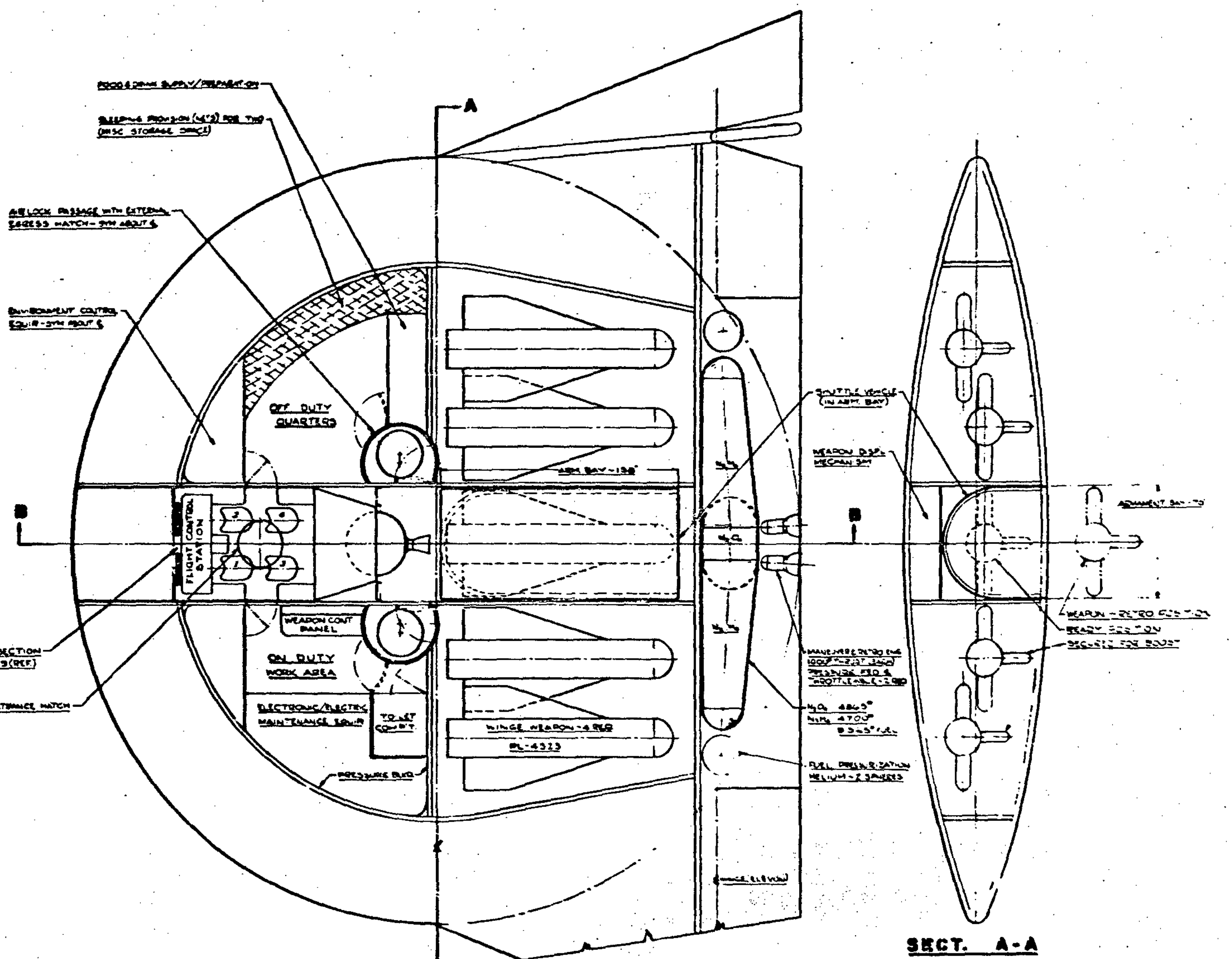


Figure 16. Manned Bombardment Vehicle Internal Arrangement

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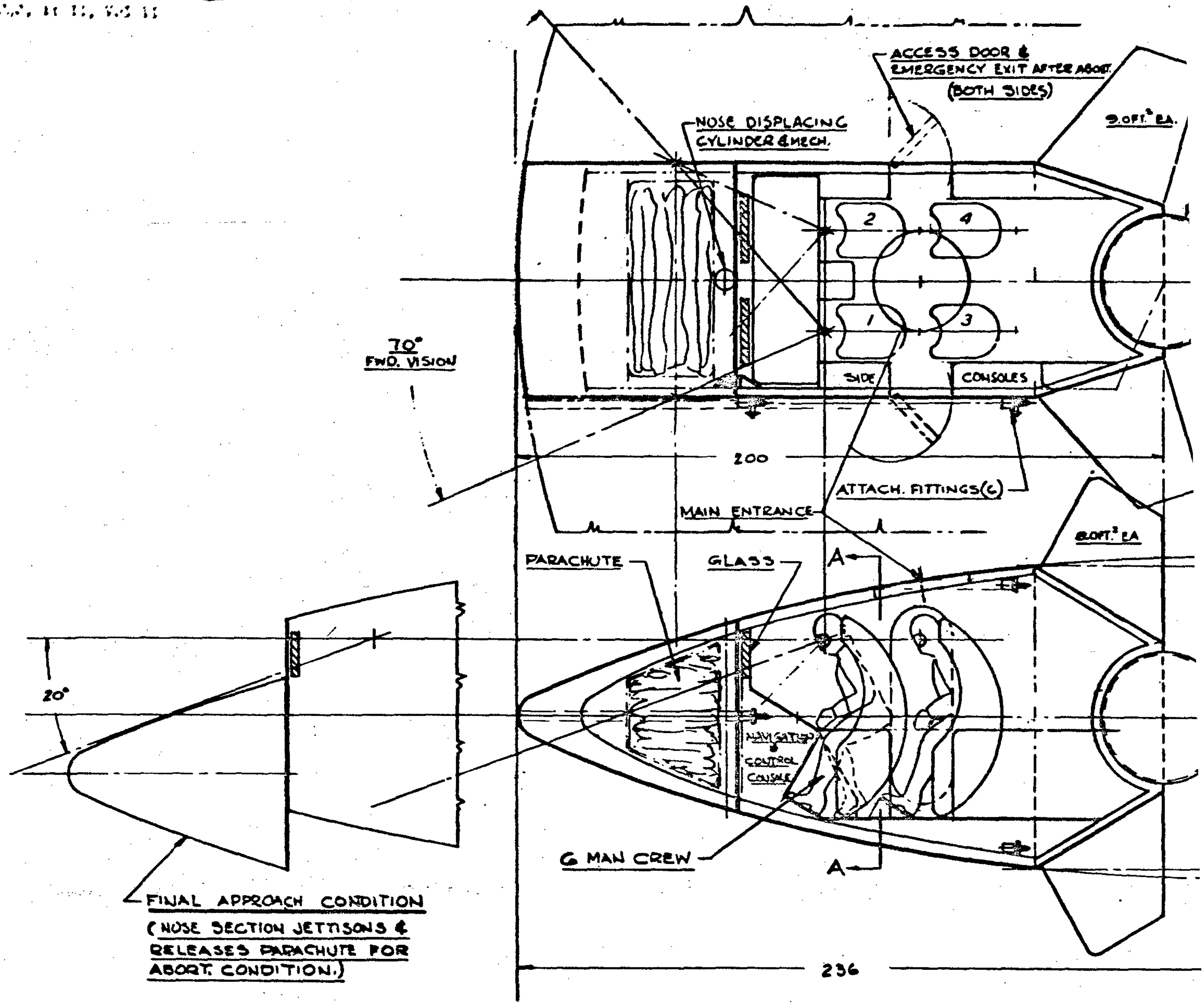


ACCESS DOOR & EMERGENCY EXIT AFTER ABORT (BOTH SIDES)
9.0FT.² EA.
NOSE DISPLACING CYLINDER & MECH.
2
4
1
3
70° FWD. VISION
SIDE
CONSOLES
200
ATTACH. FITTINGS(6)
MAIN ENTRANCE
8.0FT.² EA
PARACHUTE
GLASS
A
20°
NAVIGATION & CONTROL CONSOLE
6 MAN CREW
FINAL APPROACH CONDITION (NOSE SECTION JETTISONS & RELEASES PARACHUTE FOR ABORT CONDITION.)
236

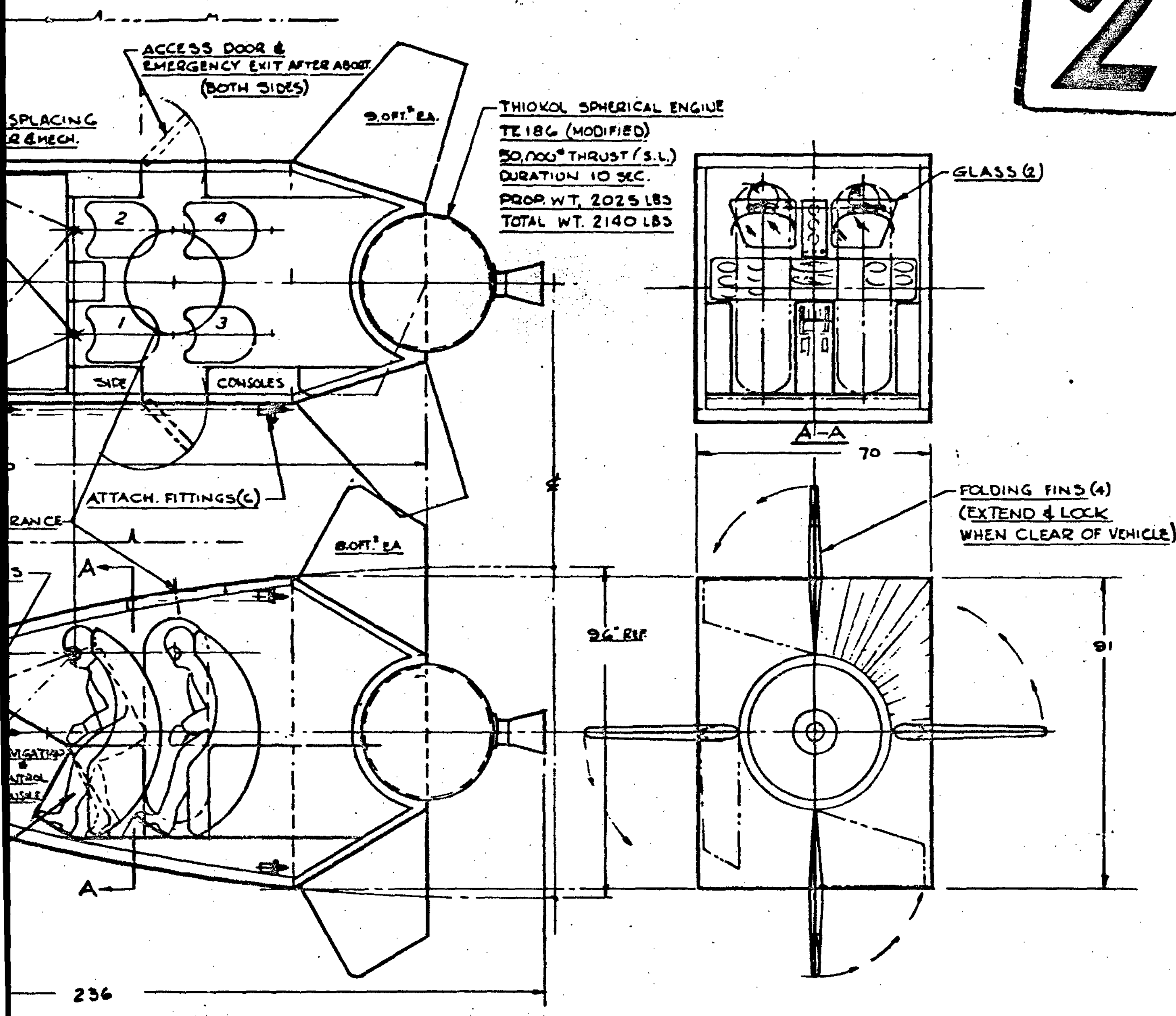


Figure 17. Crew Escape Capsule

TABLE 9.

MANNED BOMBARDMENT VEHICLE WEIGHT SUMMARY

Item	Weight (pounds)	
Structure	8,740	
Propulsion	2,602	
Fixed equipment	5,700	
Weight empty		17,042
Crew (4)	1,000	
Propellant (usable)		
Retro and maneuver	9,125	
Attitude control	250	
Trapped propellant	190	
Abort vehicle (less crew)	5,000	
Service vehicle	3,756	
Pressurizing helium	51	
Water	330	
Weapons (4)	8,056	
Tools and spare parts	200	
Useful load		27,958
GROSS LAUNCH WEIGHT		45,000
Propellant	9,375	
Service vehicle propellant	1,900	
Water	330	
Expendables		11,605
LANDING WEIGHT		33,395

in Table 10. The capsule separates the crew living quarters from the work area and provides access to either compartment through a door in each side of the capsule. A sealant material around the doors prevents leakage of the cabin air during normal operation and is broken only when the capsule is separated from the primary vehicle. The abort capsule contains the crew stations for boost and reentry, the vehicle control consoles, and the emergency power supply and life support equipment needed for an abort mission. Exit from the capsule after an abort is provided by the doors on each side which serve as the passageway doors between the compartments in normal operation. Separation of the capsule from the main bomber vehicle is accomplished with a 50,000-pound thrust, spherical, solid propellant rocket engine located on the aft section of the escape capsule. The engine provides a burning duration of about 10 seconds. The abort engine imparts an initial acceleration of approximately 8.5 g to the capsule which is sufficient to escape the 5-psi overpressure wave resulting from an explosion in the first stage booster during launch from the pad. The 10-second burning duration ensures sufficient altitude above the pad at burnout to permit recovery of the capsule with a parachute located in the forward nose section of the capsule, which is enclosed with a jettisonable nose fairing. The recovery chute is sized to limit the impact velocity of the capsule to approximately 25 feet per second.

Stabilization of the capsule is accomplished with four foldout fins located on the aft section. Over-nose vision for a normal vehicle landing is provided by translating the nose section downward and exposing a flat plate windshield in the forward pressure bulkhead of the capsule. Utilization of this window for observation from the crew compartment during the orbital mission is also permissible. The environment of the crew capsule, living quarters area, and working area compartments is maintained for "shirt-sleeve" operation during the orbital mission; however, all compartments contain sufficient volume for crew operation in space suits.

Off-Duty Area

The off-duty area is the living quarters for the crew and is located on the starboard side of the vehicle. This area is designed with sufficient volume and flow area to provide comfortable living quarters and recreational area for the crew during the extended mission duration of 6 weeks. The compartment contains sleeping and sanitation provisions, food storage and separation facilities, storable table, chairs, and exercise equipment. Sleeping nets with storage space below for

TABLE 10
ESCAPE CAPSULE WEIGHT SUMMARY

Item	Weight (pounds)	
Structure	1886	
Propulsion	109	
Fixed equipment	920	
Weight empty		2915
Crew (4)	1000	
Propellant	2085	
Useful load		3085
GROSS LAUNCH WEIGHT		6000
Propellant	2085	
Expendables		2085
LANDING WEIGHT		3915

personal gear, books, etc., are located along the wall. Food storage and preparation facilities, plus miscellaneous storage provisions, are located along the other wall. An air lock compartment provides access from the pressurized off-duty area to the unpressurized weapons bay, or to the outside of the vehicle. Access to the on-duty work area is through the vehicle control compartment.

On-Duty Area

The on-duty work area is located on the opposite side of the vehicle. The work area contains all the display consoles and control equipment for launching, monitoring, and controlling the weapons and controlling the unmanned weapon clusters, the primary electronic equipment, and miscellaneous tools and maintenance equipment. A generous allowance of volume for the display consoles and electronic weapon control equipment permits easy access to all the equipment for maintenance and repair. The primary environmental control, power supply and miscellaneous vehicle support equipment is located around the forward periphery of the off-duty and on-duty compartments. The central floor area of the work area is kept open to permit freedom of movement for the crew for maximum efficiency and convenience. Access to the outside and to the weapon storage area is provided with an inter-connecting air lock compartment.

Weapon Storage Area

The weapon storage compartment is located in the aft section of the vehicle on each side of the weapon launch bay and provides storage space for four winged weapons - two on each side of the displacing mechanism which is located on the lift coefficient of the vehicle. The weapon storage area is an unpressurized compartment and is accessible from both the on-duty and off-duty crew compartments through air lock compartments.

Sufficient space is provided for access to the weapons in the storage area for maintenance, repair, checkout, and launch procedures. The weapons are supported on rails in the storage area which permit manual transfer of the weapons from the stored position to the displacing mechanism for launch. During the orbital phase of the mission, all weapons will be transferred, via the displacing mechanism, after the shuttle vehicle is removed, and attached to the external surface of the vehicle in ready position. The shuttle is then returned to the displacing mechanism and the vehicle is ready to initiate a strike using the attached weapons or using remotely located clustered weapons. The vehicle is also in position to effect an immediate reentry, in case of emergency, by simply detaching the four weapons and leaving them in space to be recovered at a later time.

An alternate use of the weapon storage area will be as a logistic cargo compartment for transporting fuel and replacement parts as needed for service and repair of the unmanned weapon clusters.

Flight Performance

Aerodynamics

Primarily because of its excellent surface area-volume-weight relationship, the lenticular shape has been chosen as its satellite-reentry configuration for the manned bomber. The basic disc shape is inherently unstable assuming a representative center of gravity location. However, control surfaces, flaps, and speed brakes suitably located and configurations tailoring can make the lenticular shape stable and, with other desirable characteristics, a very satisfactory manned reentry and landing configuration will evolve. A Global Surveillance System using this same basic lenticular concept is described in Reference 7.

Subsonic L/D_{max} is approximately 9, which will result in excellent landing characteristics. The L/D_{max} at supersonic speed is about 2.0, and at hypersonic speeds this value would decrease to about 1.5 which is ample for reentry maneuvering.

Entry Performance

For all the entry trajectories, it was assumed that the maximum lift coefficient capabilities of the vehicle was 0.70 occurring at an angle of attack of 51 degrees. The maximum L/D of 1.5 occurs at $\alpha = 14$ degrees which corresponded to a lift coefficient of 0.28. Four basic entry trajectories are presented in Figures 18 through 21. In Figure 18 a manned entry is shown initiated by a retrograde impulse of 200 feet per second below circular orbit speed at 400,000 feet. The entry is flown at a constant maximum lift-coefficient and results in the coolest leading-edge temperatures of those trajectories studied. Two alternate entries are shown in Figures 19 and 20. These trajectories were initiated with similar conditions, but lift is arbitrarily varied to represent a maneuvering entry. Leading-edge temperatures encountered during these entries are considerably higher and have been accounted for in the structural design. Figure 21 presents an entry initiated by a 500-foot-per-second retrograde impulse in order that entry time may be reduced. This may be desirable due to system malfunctions or solar flare and radiation warnings.

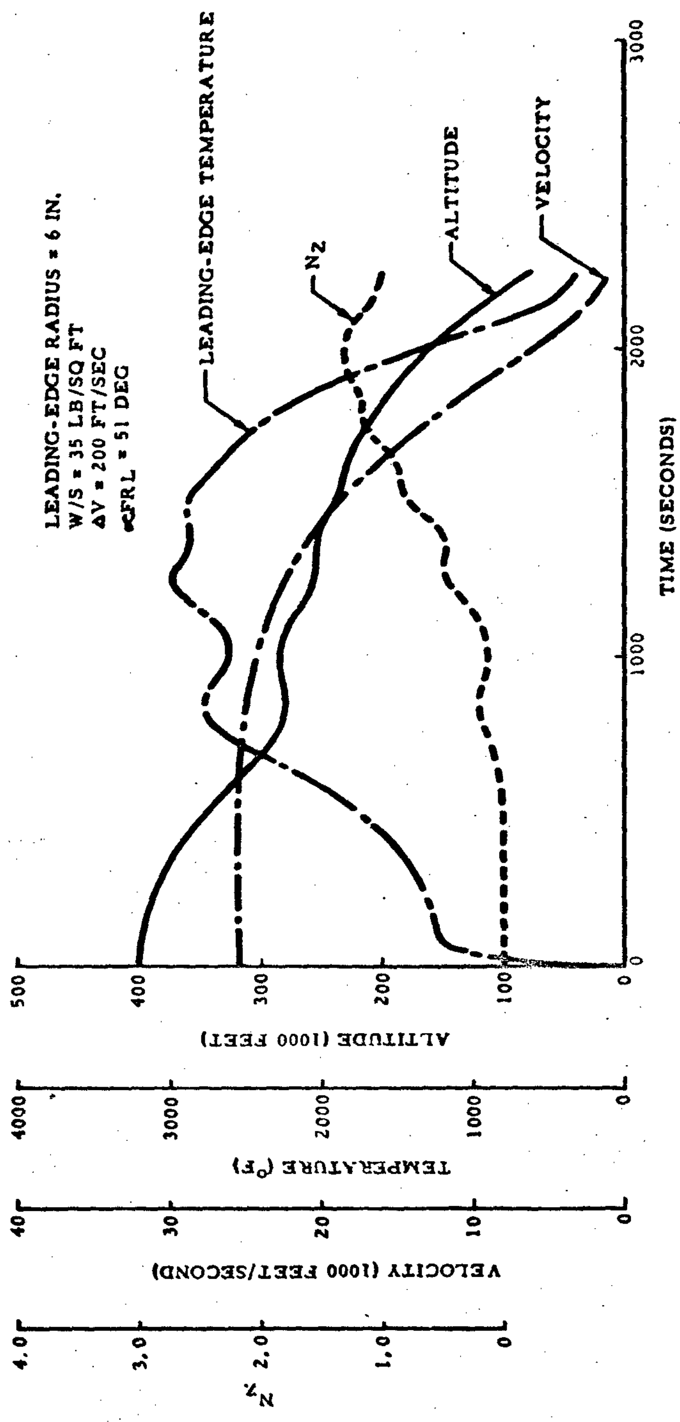


Figure 18. Reentry From Orbit ($C_{L_{max}} = 0.7$)

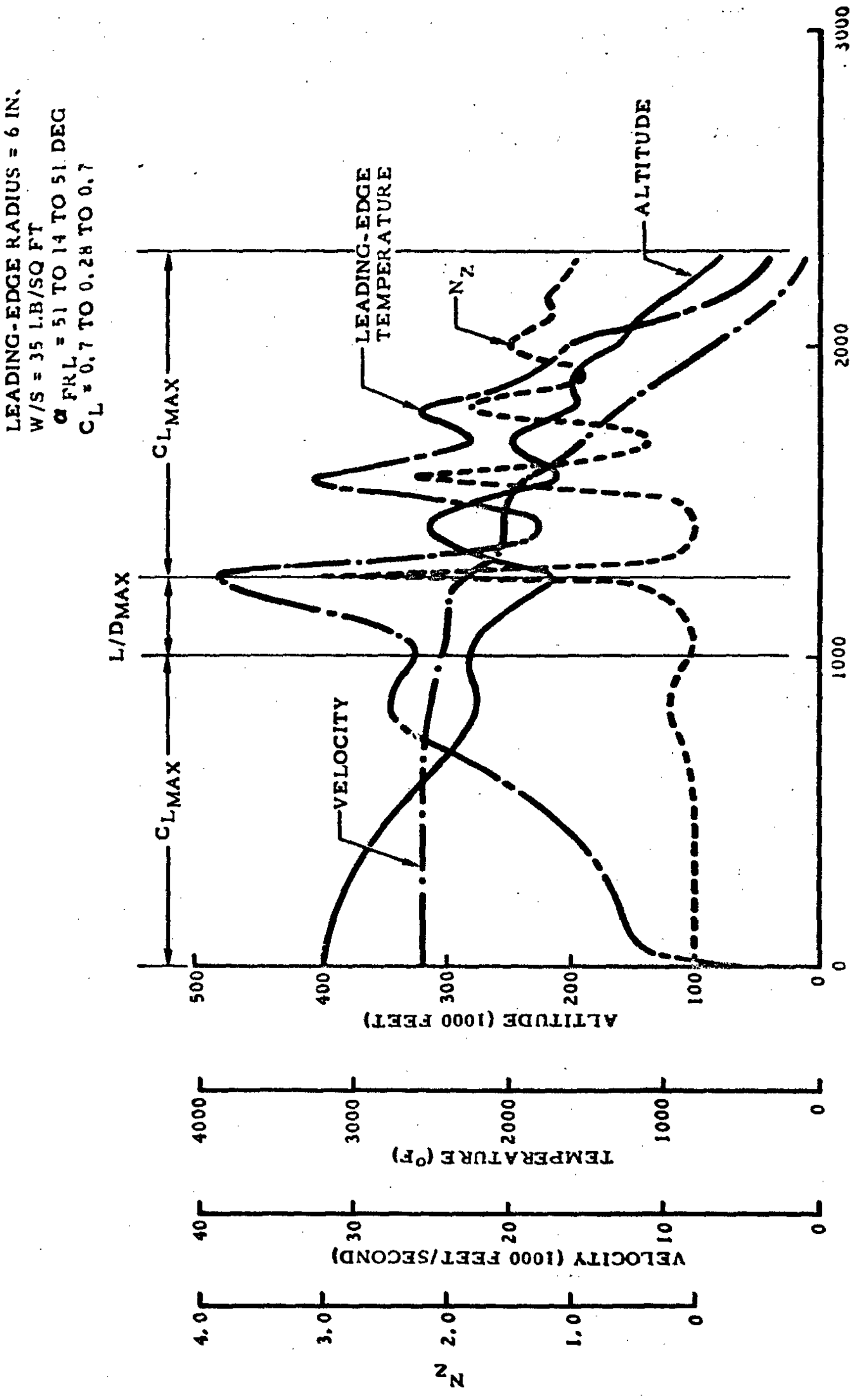


Figure 19. Reentry From Orbit ($C_{L_{max}}$ = L/ D_{max} to $C_{L_{max}}$)

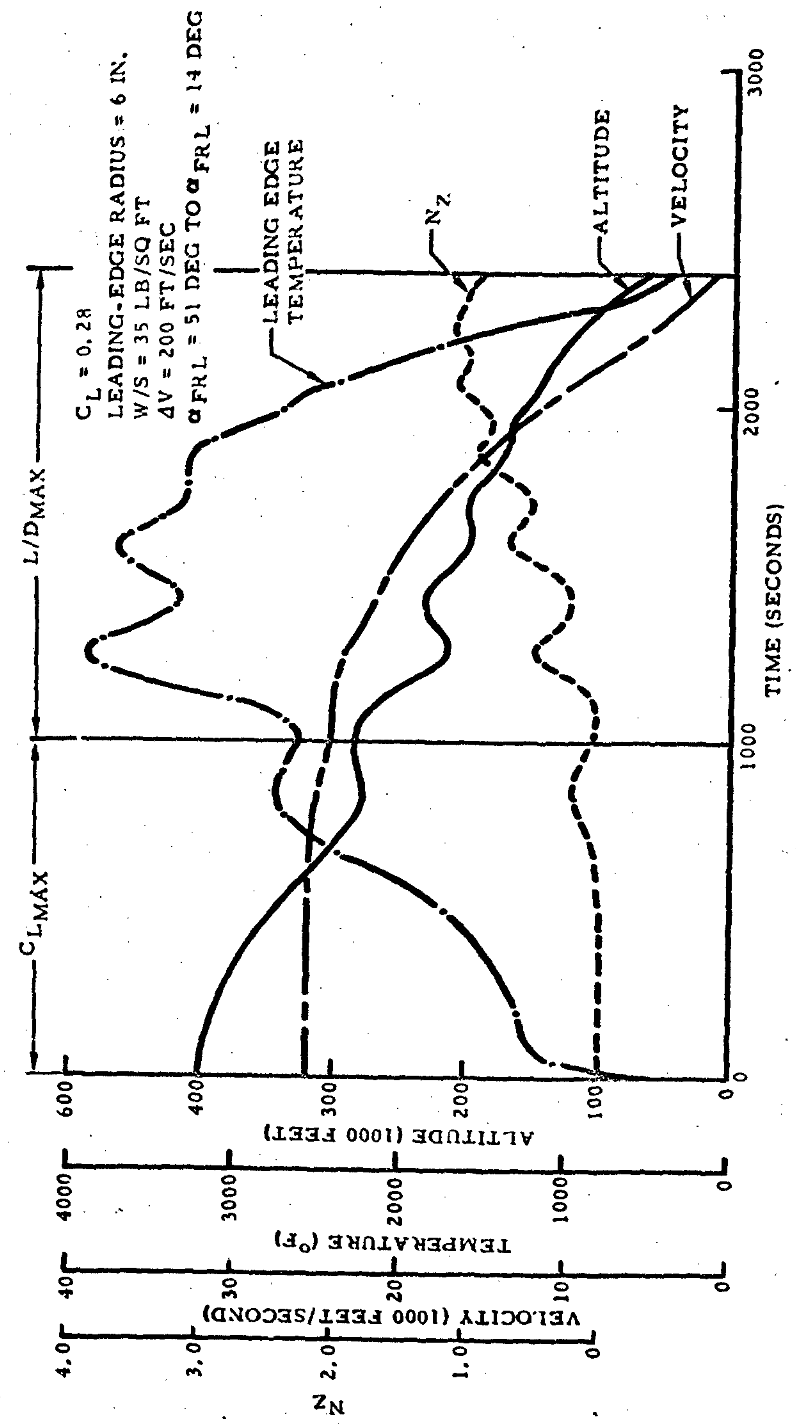


Figure 20. Reentry From Orbit ($C_{L_{max}}$ = 0.7 L/ D_{max})

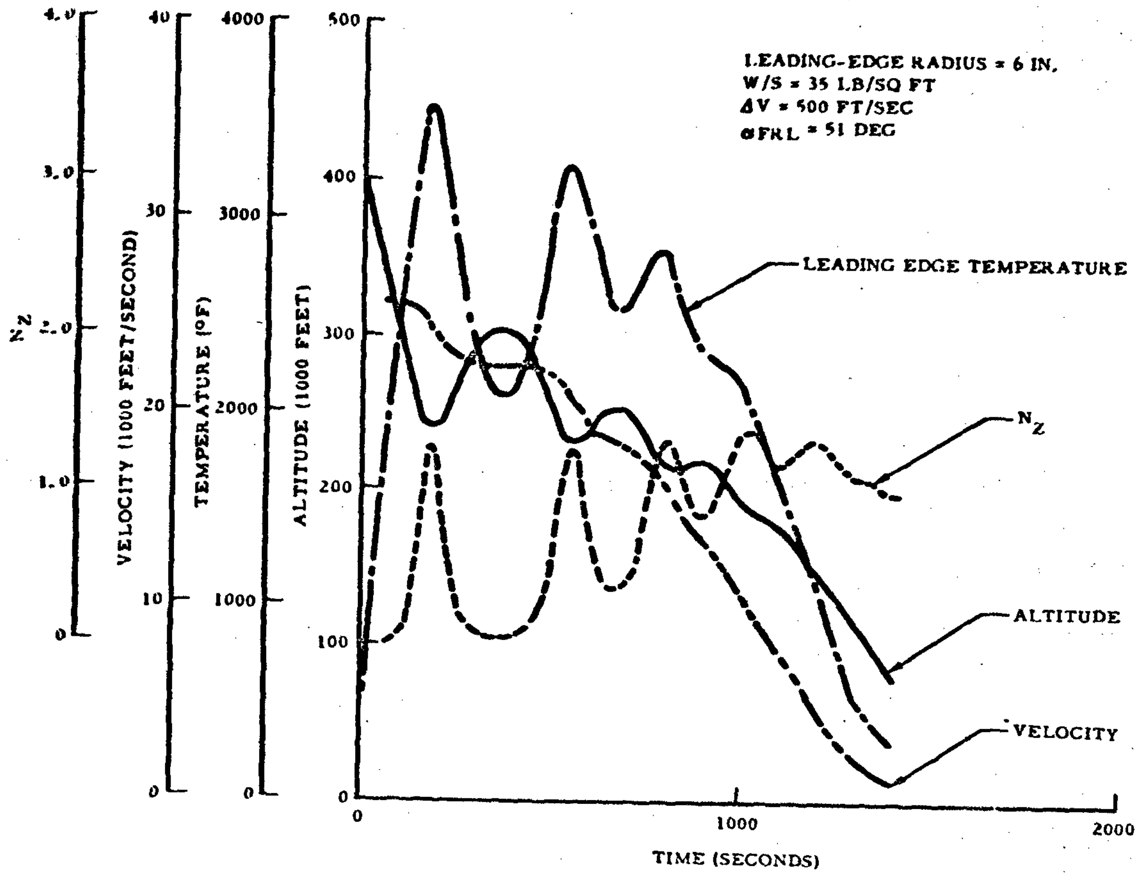


Figure 21. Reentry From Orbit ($C_{L_{max}}$ = 0.7; 500 Feet Per Second ΔV)

Thermodynamics

The lenticular-shaped reentry vehicle is favorable from a heating standpoint, when compared to other reentry vehicle configurations. The leading edge of the disc may be treated as a cylinder normal to the flow, which results in two-dimensional flow is a reduction of the aerodynamic heating rate to the leading edge of $1/\sqrt{2}$ or 0.707 of the heating rate that would normally reach a hemisphere or blunted cylinder during a similar entry mission.

The disc-shaped configuration with control surfaces on the aft portion of the vehicle eliminates the problem of high heating due to low shock interactions between conventional fuselage nose and wing leading-edge surfaces. This problem is common to winged body lifting vehicles.

Another advantage of the disc configuration is that large leading-edge radii may be achieved. This results in a reduction of the aerodynamic heating rate that would be experienced by thin-winged entry vehicles.

Structural Design

The manned bombardment vehicle must be capable of supporting its flight and dynamic loadings throughout the mission profile without sustaining permanent damage. The mission profile of this vehicle starts at the launching pad, continues through 6 weeks in orbit, through reentering the earth's atmosphere, and concludes with maneuvering to a landing. After each mission, the airframe will require visual inspection to detect any meteoroid damage or surface erosion. These minor damages should be the only airframe service required prior to the next flight.

In order to provide a reliable vehicle for the accomplishment of the above mission the structural design must follow a rigorous program involving criteria, materials, and structural arrangement. The criteria is studied in four phases that are consistent with the operational mission of the vehicle.

Ground Launch — A maximum axial load factor of NX = 8.0 g (limit) is used. This load factor also accommodates engine thrust overshoot.

The structural temperature is unspecified, but is assumed to be less critical than the reentry temperature. The above load factor is applied to the vehicle by the interstage structure through the main fore and aft longitudinal beams that are adjacent to the personnel escape module. These beam caps are reinforced locally at the junction to the interstage and the webs also serve as pressure isolation members in the vehicle cabin area.

Wind shears associated with the boost phase induce the maximum bending moment in the interstage structure.

Orbital Flight — During this phase of the mission there are essentially no loads on the vehicle other than control thrusts. These loads are so small that the structure, other than the immediate attachment, is designed by some other criteria. The primary structural problem during orbital flight is the provision of a non-leaking pressure-tight cabin.

Reentry — Preliminary reentry trajectory studies, including the effects of vehicle geometry, showed that a leading-edge radius of 6 inches was sufficient to keep the temperature within the limits of coated graphite. Using this radius, time-temperature-load factor profiles were established for several different reentry trajectories. The limits of these trajectories were the maneuvering range from $C_{L\ max}$ to L/D_{max}. The $C_{L\ max}$ reentry generates minimum structural temperatures, but does not allow the maneuverability of lower C_L profiles. Emergency reentry, utilizing maximum retro thrust, generates higher temperatures which are compatible with L/D_{max} reentry temperatures. Maneuvering from L/D_{max} to $C_{L\ max}$ provides design points for combinations of high temperature and high loads. The reentry profile for this condition is shown in Figure 19 and the resulting fuselage lower surface and upper surface temperature profiles are shown in Figures 22 and 23, respectively.

The cabin is pressurized to 10 psi limit continuously from 10,000 feet altitude in the boost phase through the 6-week orbiting flight and reentry back to 10,000 feet altitude.

Landing — The normal landing load factor is limited to 2.0 g limit by pneumatic landing gear struts.

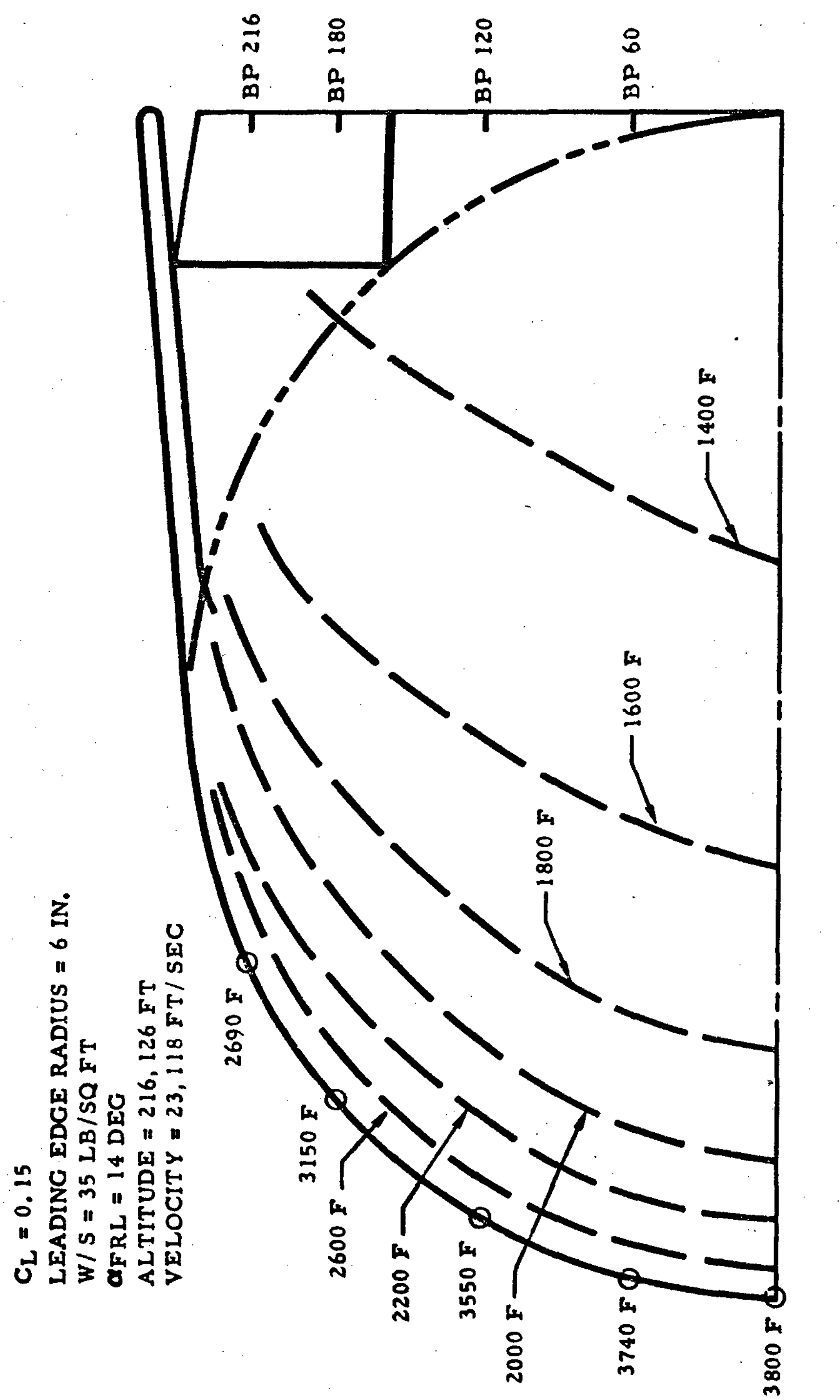


Figure 22. Lower Surface Temperature

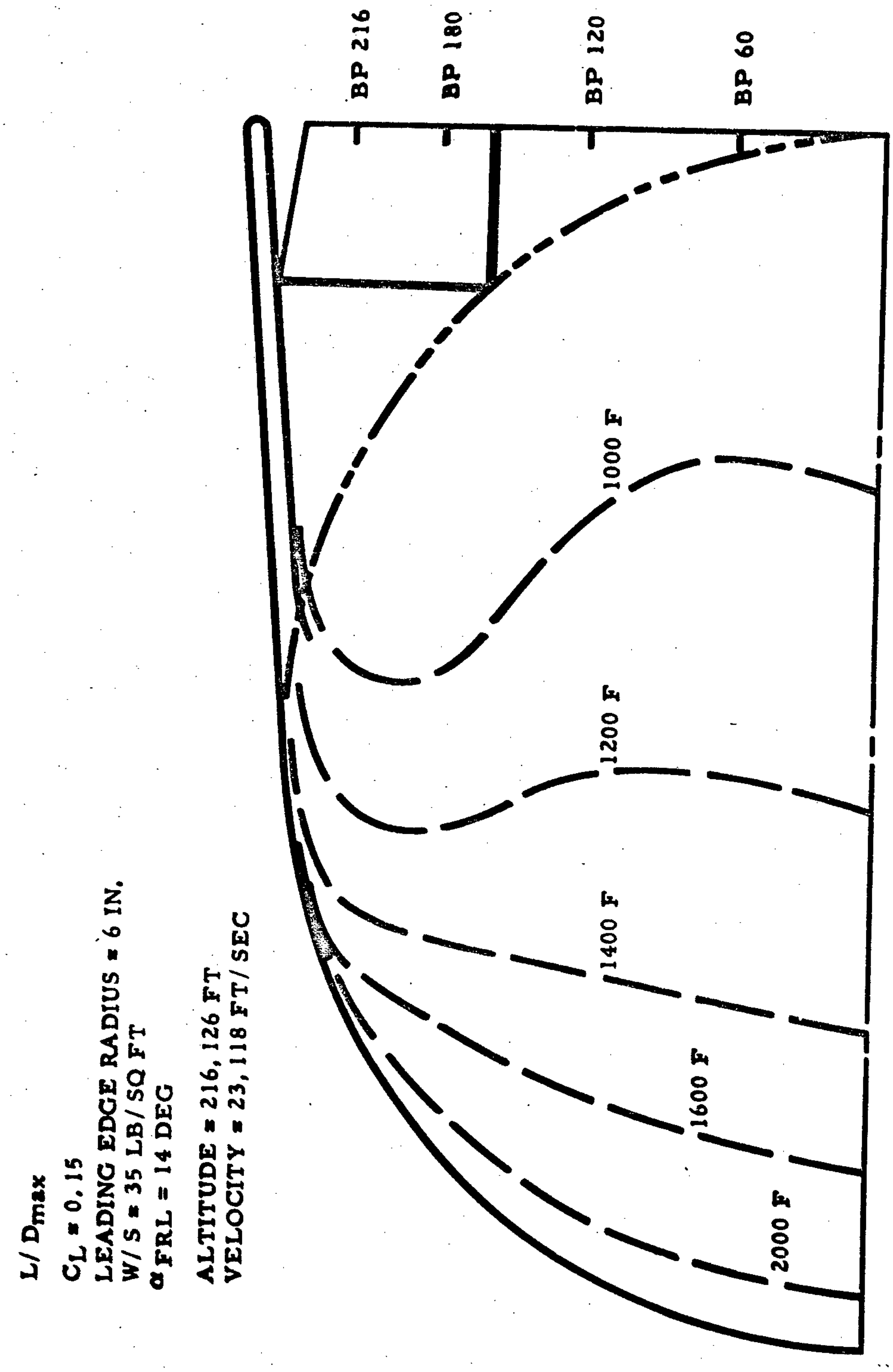


Figure 23. Upper Surface Temperature

Propulsion Systems

There are three basic propulsion systems contained within the manned bomber; the basic vehicle maneuvering system, the shuttle vehicle maneuvering system, and the crew abort system. The crew abort system is a single-purpose system while the manned vehicle maneuvering system is integrated with the shuttle vehicle system, as well as with the propulsion systems contained in the unmanned weapon carrier or cluster, and in the weapons themselves.

All of these systems (manned vehicle, shuttle vehicle, clusters, and weapons) utilize the same storable, hypergolic propellant combination; nitrogen tetroxide and hydrazine, N_2O_4/N_2H_4. The main propellant storage is in the manned vehicle, which has a capacity for 9375 pounds of propellant. The shuttle vehicle capacity is 1900 pounds, while the cluster and weapon (each) capacities are 700 pounds and 200 pounds, respectively.

Because the manned vehicle contains the majority of the propellants the shuttle vehicle will be refueled from this main supply, as required. The shuttle vehicle, in turn, will top-off the supply in each cluster as it makes its periodic servicing trips. The cluster fuel supply is continuously connected to the propellant tanks of each weapon in the cluster and thus maintains the proper fuel level. If a weapon is fired at a target, or otherwise disconnected from the cluster, automatic check valves will prevent fuel leakage.

Top-off propellant requirements for the cluster and for the weapons will be small; estimated to be less than 50 pounds per complete cluster per 6 weeks period. The ratio of absorptivity to emissivity of the external surface of either spherical or cylindrical tankage can readily be established such that, in the spatial environment, the propellants can be maintained between limits of +40 and +70F. In the case of spherical tankage this ratio will be between 1.0 and 1.4, while in the case of cylindrical tankage it will be between 4.5 and 7.0.

The basic vehicle maneuvering system is normally used to establish precision orbits immediately after injection, and to provide retrograde impulse prior to reentry. Alternate uses of the maneuvering system will include a limited orbital maneuver capacility for the manned vehicle which can be used to correct its ephemeris after long period drift, or can provide maneuvering functions which may be required as a result of special situations, such as a rendezvous requirement.

This engine provides a nominal 2000 pounds of thrust at a very low chamber pressure of about 60 psi. Recent tests indicate that throttling to 5 percent of nominal thrust will not be adversely affected by this low pressure since stable burning can be attained at chamber pressures as low as 1 psi under vacuum conditions. This engine will be radiation cooled and will provide unlimited restart capability. The expansion area ratio will be about 40:1 which will give very good specific impulse characteristics as well as providing extensive area for cooling.

The propulsion system required for the shuttle vehicle develops a nominal thrust of 200 pounds. Detail characteristics of this engine are identical to those of the manned vehicle system except for thrust level.

The abort engine is a conventionally designed, spherical, solid propellant system producing 50,000-pound thrust at sea level. This is a "one-shot" non-reusable system with a compromise nozzle area ratio of about 8:1, because abort must be performed throughout the entire altitude range of the earth boost trajectory.

Secondary Power

The manned bomber requires two separate power systems; one for the boost and reentry phases and another for the normal 6-week orbital operation. Unfortunately, it is not feasible to provide one system which can supply the energy for both requirements.

Energy for the orbital operation can most feasibly be supplied from nuclear or solar sources because of the long flight duration and relatively high continuous power level required. Approximately 7 kilowatts will be required continuously. Neither of these systems are suitable for the boost and reentry phases.

The nuclear reactor cannot be activated until the vehicle is in orbit, and on reentry, would probably be left in space to avoid the possible hazards associated with a hot reactor should a crash occur on landing. Similarly, the solar collector of a solar turboelectric system could not be erected until in orbit, and would have to be dismantled or discarded prior to reentry. Although it might be possible to devise a heat storage reservoir of sufficient capacity with the solar system to provide boost and reentry power, this could not reasonably be placed in the control vehicle because of weight and volumetric penalties.

Orbital Power

The manned bomber will use a solar turboelectric system for its orbital phase because of present nuclear reactor shielding problems. The total weight will not exceed 800 pounds including a 27-foot diameter, hinged solar collector-radiator. This estimate is based on a 90-minute orbit with a mean 35-minute dark period.

The solar collector will be similar in construction to the SUNFLOWER design in that ti will consist of hinged petals which can be expanded after an orbit is achieved. However, it will also have a 500 F radiator on the back side to which a circulating fluid will carry waste heat to be radiated into space. Orientation of the collector toward the sun within ±0.75 degree is accomplished by a reaction jet attitude control system and a sun-seeking system which are integral with the collector itself.

Light from the collector will be focused on a cavity receiver in which a lithium hydride storage unit is interposed between the interior absorbing surface and mercury boiler tubes. This storage unit will have sufficient capacity to produce the required energy for the vehicle during the dark periods of the orbits.

A binary Rankine-cycle conversion system will be utilized to change the solar heat into mechanical energy because it has high efficiency with minimum weight and size. Mercury and steam are two possible fluids for the heat exchangers.

The secondary fluid could lubricate the bearings on the hermetically sealed package which contains the primary and secondary turbines, the alternator and the pump, all of which rotate on the same shaft. Although the maximum operating temperature of this system will be 1200 F, the alternator and bearings will be subjected to a maximum temperature of 250 F.

Included in the heat engine will be a compact mercury vapor condenser, a secondary fluid boiler, a steam condenser and assorted piping, fittings, and supports.

A 7-kilowatt, 1000-csp, three-phase, 110-volt per phase alternator will transform the turbine power into electric power. Voltage and speed control regulating units will automatically maintain voltages and turbine speeds within prescribed limits.

Boost and Reentry Auxiliary Power

A silver-oxide zinc cell battery will supply the peak reentry load of 12 kilowatts for 10 minutes plus an average load of 7 kilowatts for a maximum time of 2 hours. This battery will weigh approximately 200 pounds and have a volume of 1-1/2 cubic feet. After each 6-week mission, the battery will be replaced with a new unit.

Weightwise, a cryogenic hydrogen-oxygen motor and generator would be better than a battery, and a hydrazine turbine would be as good as a battery. However, considering cryogenic storage for periods of as long as 6 weeks, a static battery system is to be preferred over a rotating turbine, on the basis of reliability and maintainability.

Installation

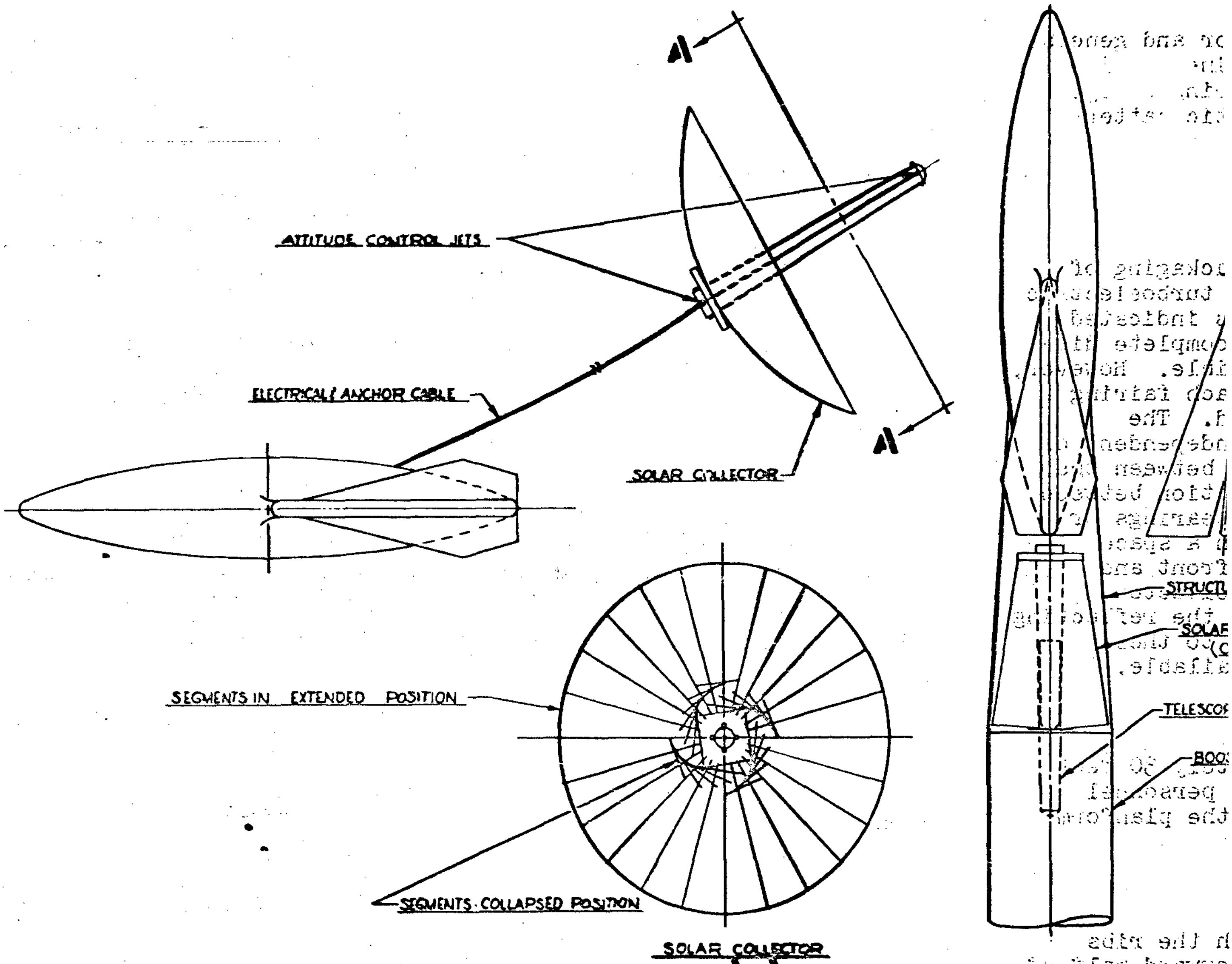
A number of difficult problems, including packaging of the array, must be solved when designing a solar turboelectric system for space. A possible packaging scheme is indicated in Figure 24. This scheme does not require the complete dismantling of the solar collector and appears feasible. However, external packaging will minimize the booster attach fairing length and may be the optimum installation method. The position and attitude of the collector will be independent of the vehicle except for the electrical connection between the collector and the vehicle. The electrical connection between the collector-radiator and the vehicle requires bearings or flexible couplings, all of which pose problems in a space environment. Temperature gradients between the front and back of the collector-radiator can distort the collector structure and degrade the optical performance of the reflecting surface. However, it is believed that solutions to these problems are possible in the development time available.

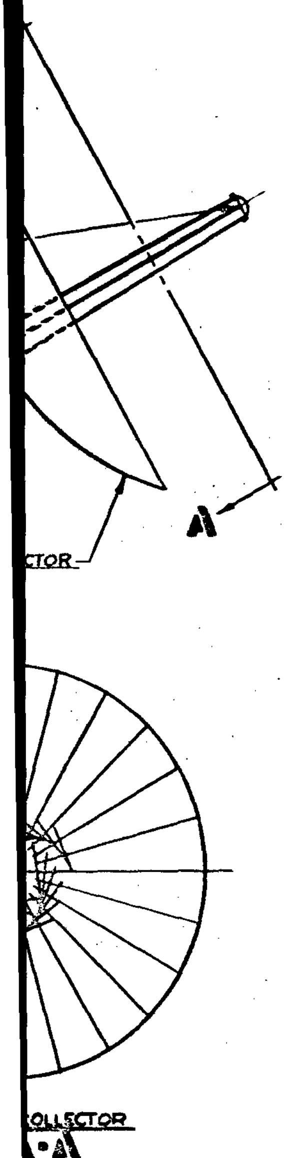
Structural Arrangement

The inner section of the vehicle (approximately 30 feet long and 30 feet wide) housing the equipment and personnel is defined as the fuselage and the remainder of the planform is defined as the wing.

Wing Structure

The wing structure is a multirib design with the ribs oriented normal to the leading edge toward the forward half of the vehicle and appearing as spars toward the rear half of the vehicle. The ribs and spars are concentrated cap design and, where needed, skin shear stiffness is achieved by adding a


ATTITUDE CONTROL JETS
ELECTRICAL & ANCHOR CABLE
SOLAR COLLECTOR
SEGMENTS IN EXTENDED POSITION
SEGMENTS COLLAPSED POSITION
SOLAR COLLECTOR
VIEW A-A



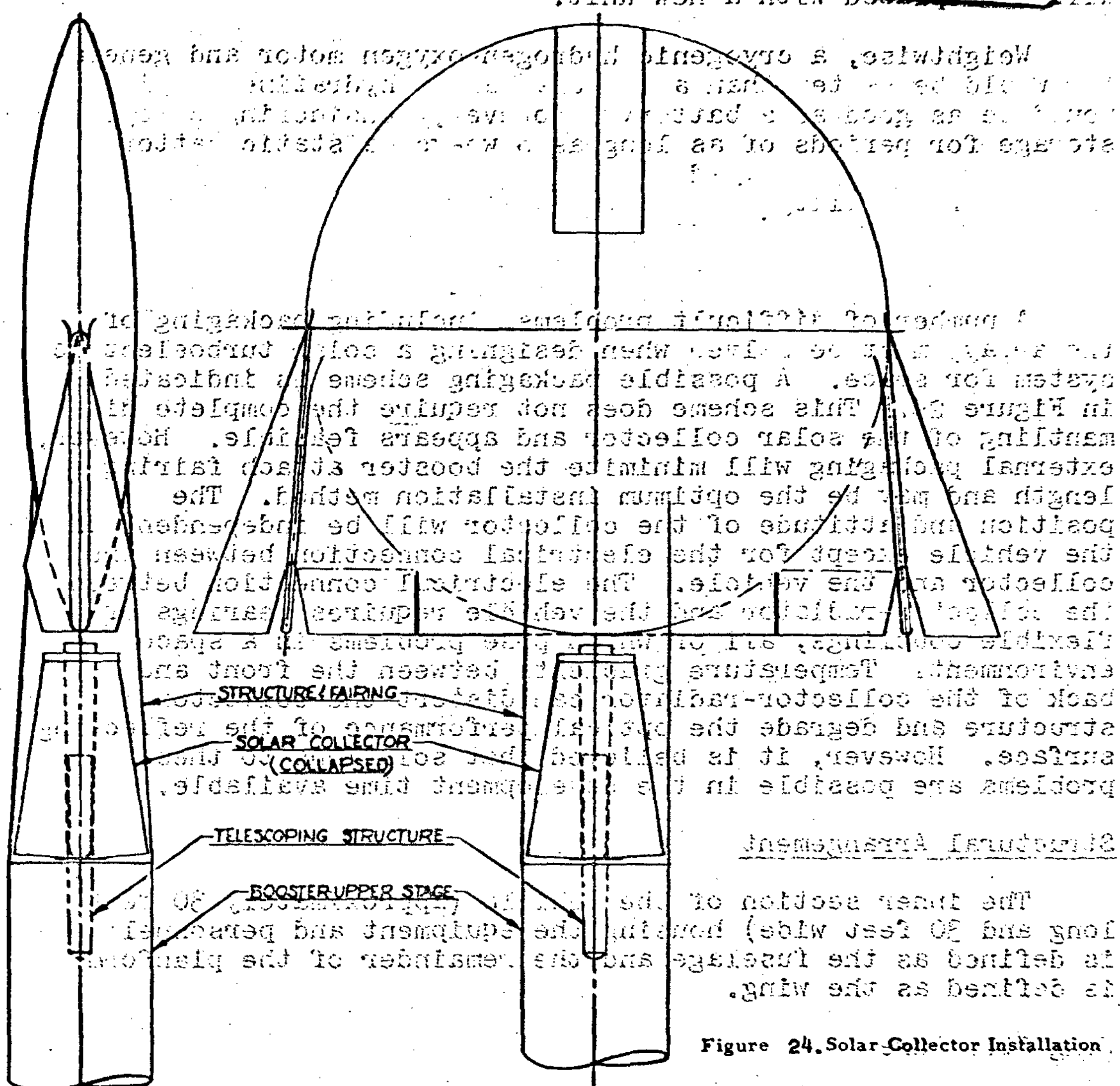


Figure 24. Solar Collector Installation

corrugated stiffening skin. The wing skins, ribs and spars are fabricated from coated columbium alloy that is allowed to heat up and reradiate. The wing structural arrangement is presented pictorially in Figures 25 and 26.

The wing leading-edge radius is 6 inches. The nose portion of the leading edge is fabricated from coated graphite and is segmented to minimize thermal strains.

Fuselage Structure

The fuselage structure is designed to function simultaneously as a load transmission structure and as a thermal protection system for the pressurized crew compartment. This structure is basically a triple wall design consisting of an outer radiation shield, a honeycomb panel primary structural shell, and an inner cabin shell. Each of these layers of the fuselage shall are separated by a layer of insulation. A typical structural arrangement of this shell is shown in Figure 27.

The outer radiation shield (reradiating surface) consists of many small independent panels that are attached to the primary shell by minimum attachment to minimize heat shorts. These panels distribute the aerodynamic loads to the primary structural shell and reach a temperature of approximately 2000 F on the vehicle lower surface and 1600 F on the vehicle upper surface.

The primary structural shell is made of honeycomb sandwich panels that are fabricated of brazed nicle base alloy (Rene' 41) face sheets and core. Panel boundaries are afforded by beams rather than tension webs to the far surface because with the more optimum panel configurations, the tension webs would render the fuselage space nearly unusable. This shell is used to react and redistribute all flight and landing loads and will, in general, be designed by cabin pressure with reinforcements and fittings as are necessitated by localized or concentrated loads. The general design of this primary shell is based on a fuselage pressure of 10.0-psi limit and an operating temperature of 1000 F. This design is shown in parametric form in the curve of Figure 28. This figure shows that the smallest feasible square panel will yield the optimum structural design. However, other criteria such as compartmentalization and equipment installation or crew movement and accessibility for service and maintenance of equipment during orbital flight necessitates a departure from optimum structural design.

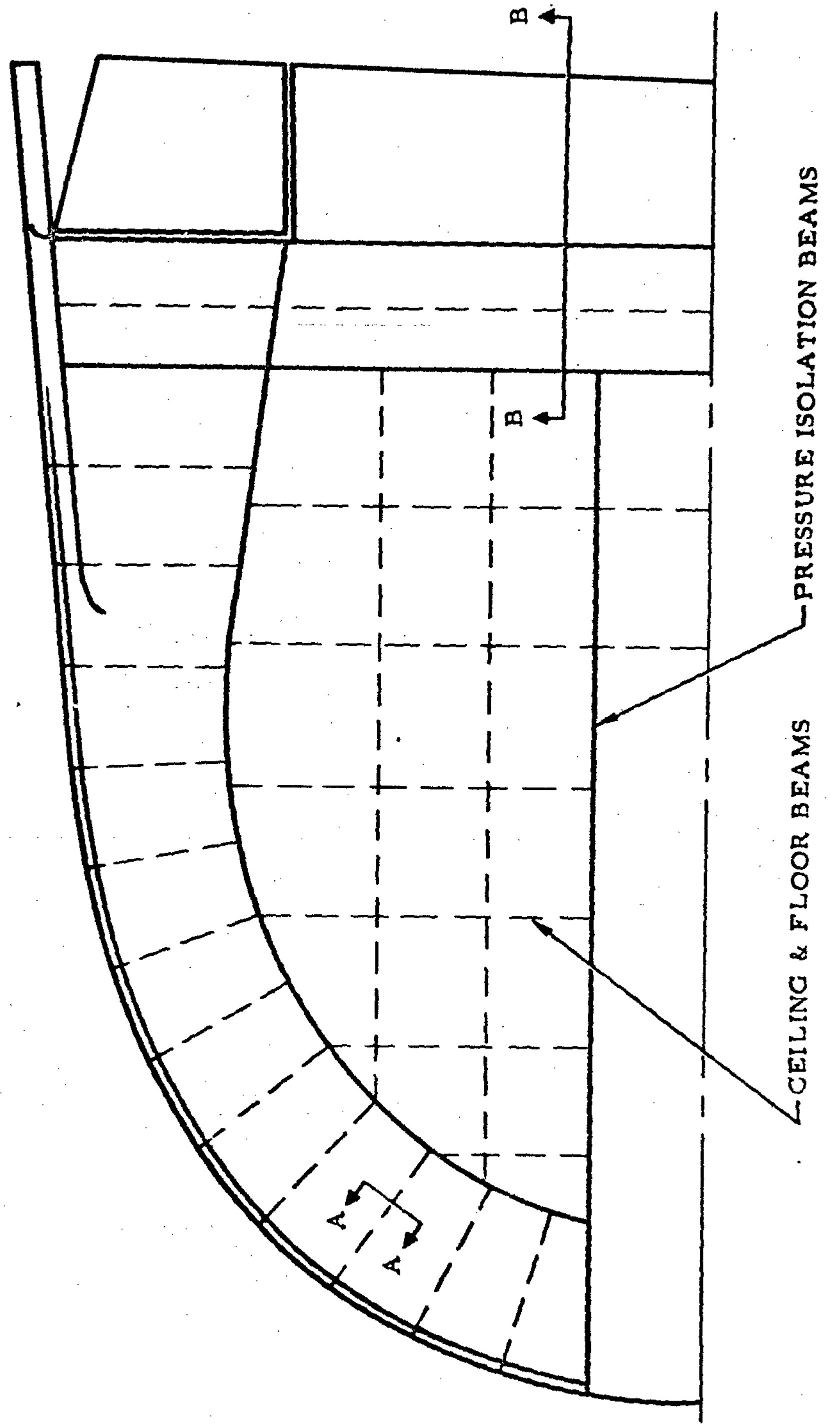
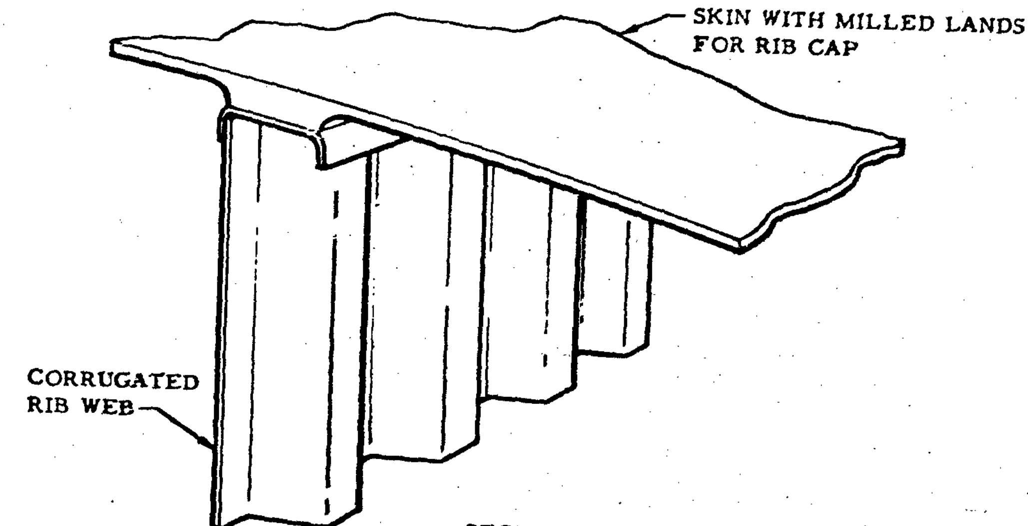
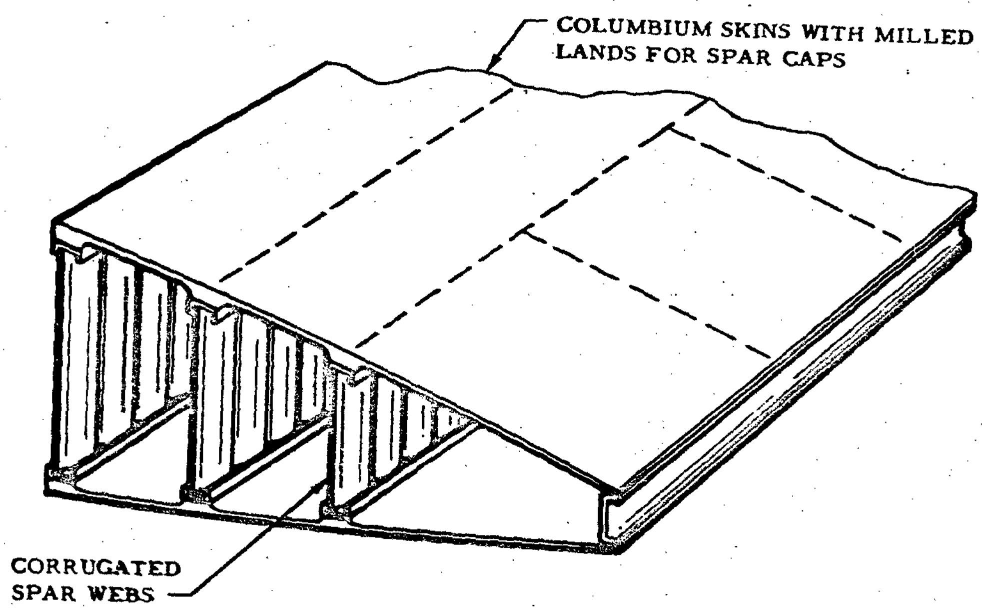


Figure 25. Wing Structural Arrangement



SECTION A-A



SECTION B-B

Figure 26. Wing Structural Details

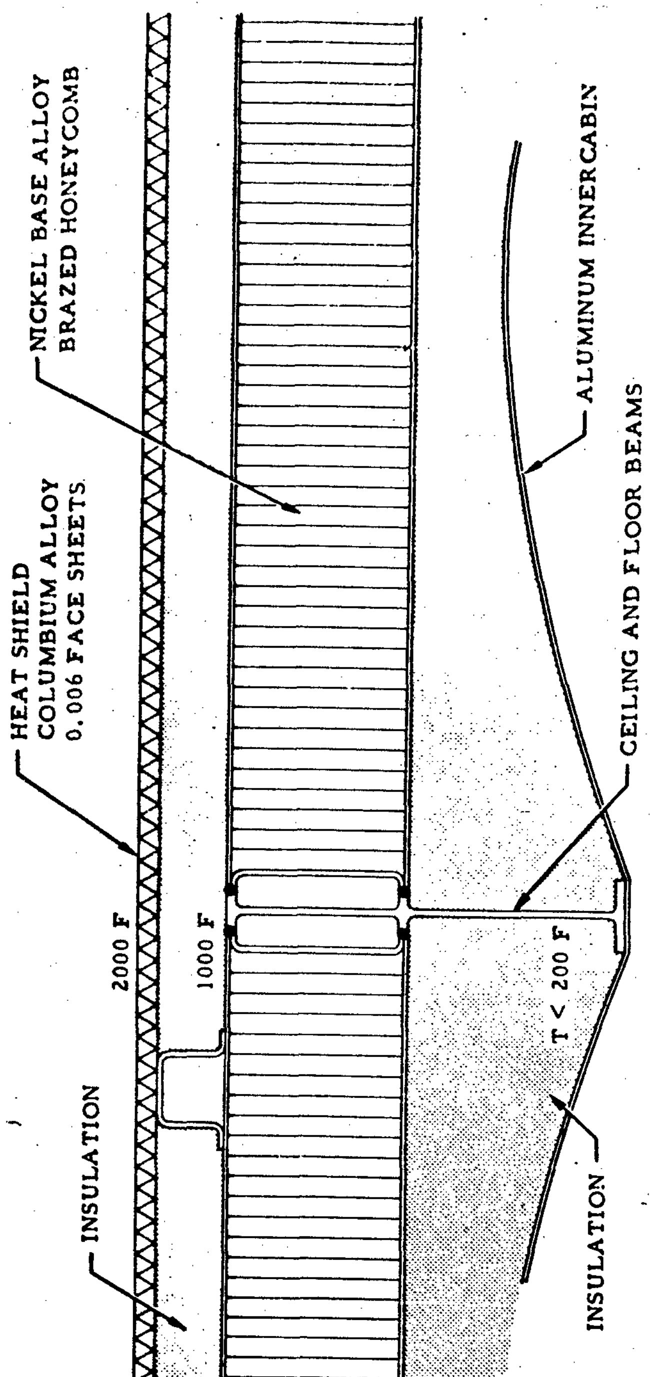


Figure 27. Fuselage Wall Construction

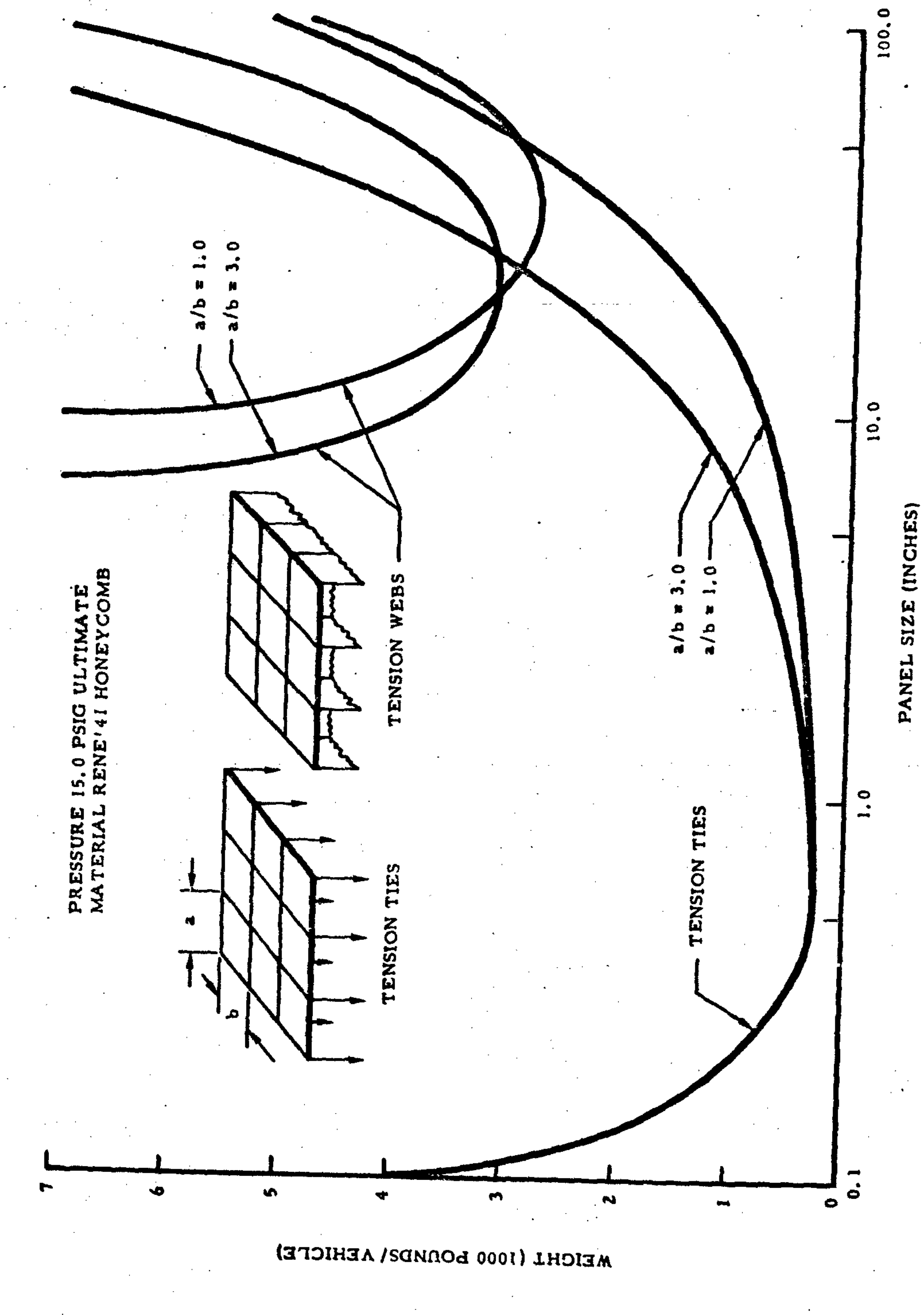


Figure 28. Fuselage Structural Shell

Thermal Shielding

Structural and environmental problems, due to the high temperatures induced during reentry, require the use of thermal shielding to control the peak temperature reached in the personnel area. In the case of wing structure, where there are no internal temperature limits, the use of shielded structure is precluded unless it proves lighter than hot radiating structure. Because the wing of this vehicle is designed as a hot reradiating structure this discussion is concerned with the fuselage portion of the vehicle.

The above problem is solved, in part, by the use of shingle-type heat shields. These shields are 24- by 4-inch rectangular sandwich panels of corrugated annealed columbium (F-48) designed to peak temperatures of about 2600 F. The face sheets are 0.006 inch thick and the shear member is a 60 degree corrugated web that is 0.003 inch thick. Between these heat shields and the primary structure is a layer of insulation which limits the temperature of the primary structure to about 1200 F. A number of heat resistant materials and alloys, among which is the nicle base alloy that was used in the design of the primary structure, have good structural efficiency in this temperature range. A layer of insulation between the primary structure and the inner cabin further reduces the temperature to something less than 200 F; depending on the type, thickness, and whether or not it is evacuated.

Because it is mandatory, due to the presence of personnel and temperature sensitive equipment, to provide a controlled temperature in the fuselage, the dependence on these parameters is studied in greater detail. The simplest and most reliable insulation is a dry-type system. If the insulation is operating in a vacuum, the thermal conductivity is approximately 1/5 of that at sea-level pressure. Studies of the reentry profile indicate that if the insulation chamber is vented to the outside, the peak temperatures occur at an altitude where the thermal conductivity is high approaching the sea level conductivity value. This makes it desirable to evacuate the insulation chambers.

Evacuation of the outer insulation compartment does not appear feasible because of the extreme temperature gradients of the outer radiating surface. Because of these high gradients, the outer surface heat shield is segmented and the accompanying numerous joints makes sealing impractical.

The above considerations for cabin areas of the vehicle leads to the conclusions that:

Inside wall maximum temperature = 200 F

Maximum primary structure temperature = 1200 F

Maximum outside skin temperature = 2600 F

Insulation chambers should be evacuated

Outside surface should be segmented

From the above conclusions, it appears to be desirable to split the required insulation, placing part of it outside of the primary structure and the balance inside the primary structure. The inboard insulation chamber can be evacuated by producing a sealed primary structure capable of supporting cabin pressure and then constructing an inner pressure vessel that forms the cabin wall. This inner vessel can be idealized for pressure loading only; the outer primary structure supports the flight and landing loads. By producing two pressure vessels one inside of the other, each capable of supporting the cabin pressure, the insulation chamber between these two vessels can be evacuated thus taking advantage of the improved insulation characteristics. This inner insulation chamber can be evacuated prior to reentry by controlled venting to space and then closing the vents for reentry.

A sketch of this type of cabin wall construction is shown in Figure 29. The curves of Figures 30 and 31 show the resulting inner wall temperature as a function of insulation thickness for this type of cabin wall construction.

Meteoroid Encounter

The foregoing paragraphs of this section have delineated the criteria and design philosophy based on the temperature and loading conditions which would normally be encountered on the specified mission. Unusual environmental situations which will develop, such as meteoroid encounter, must also be considered in the design.

Figure 32 shows the probability of at least one meteoroid encounter, as a function of meteoroid diameter. For example, it is noted that an encounter with a meteoroid of 0.12 inch diameter would be anticipated with a 0.88 probability.

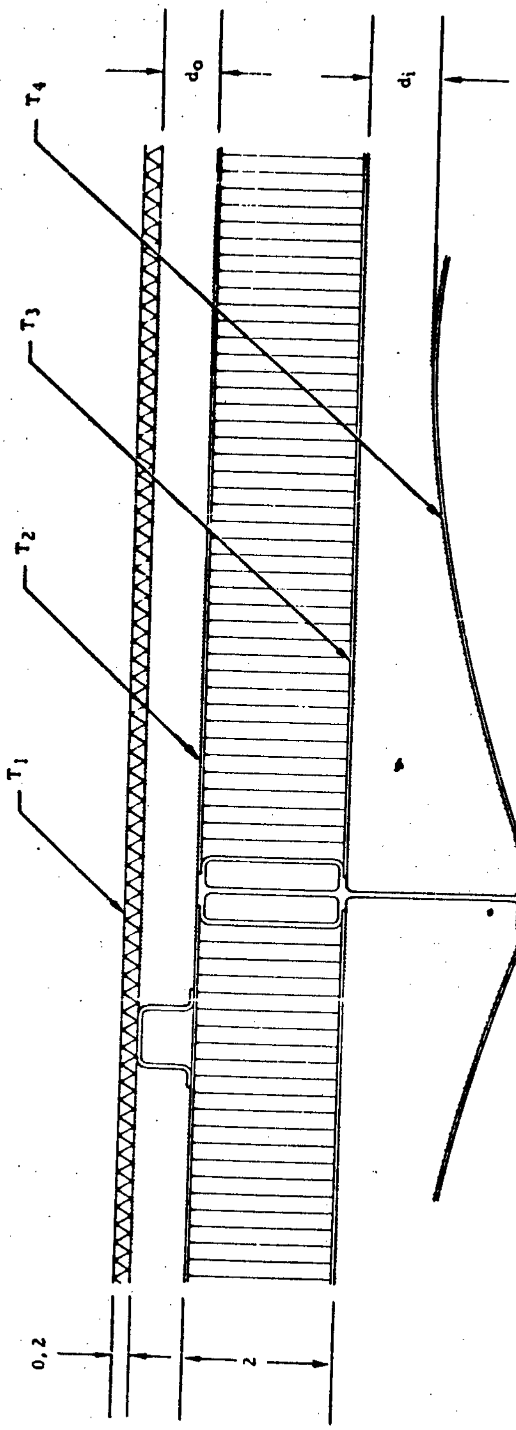


Figure 29. Cabin Wall Construction

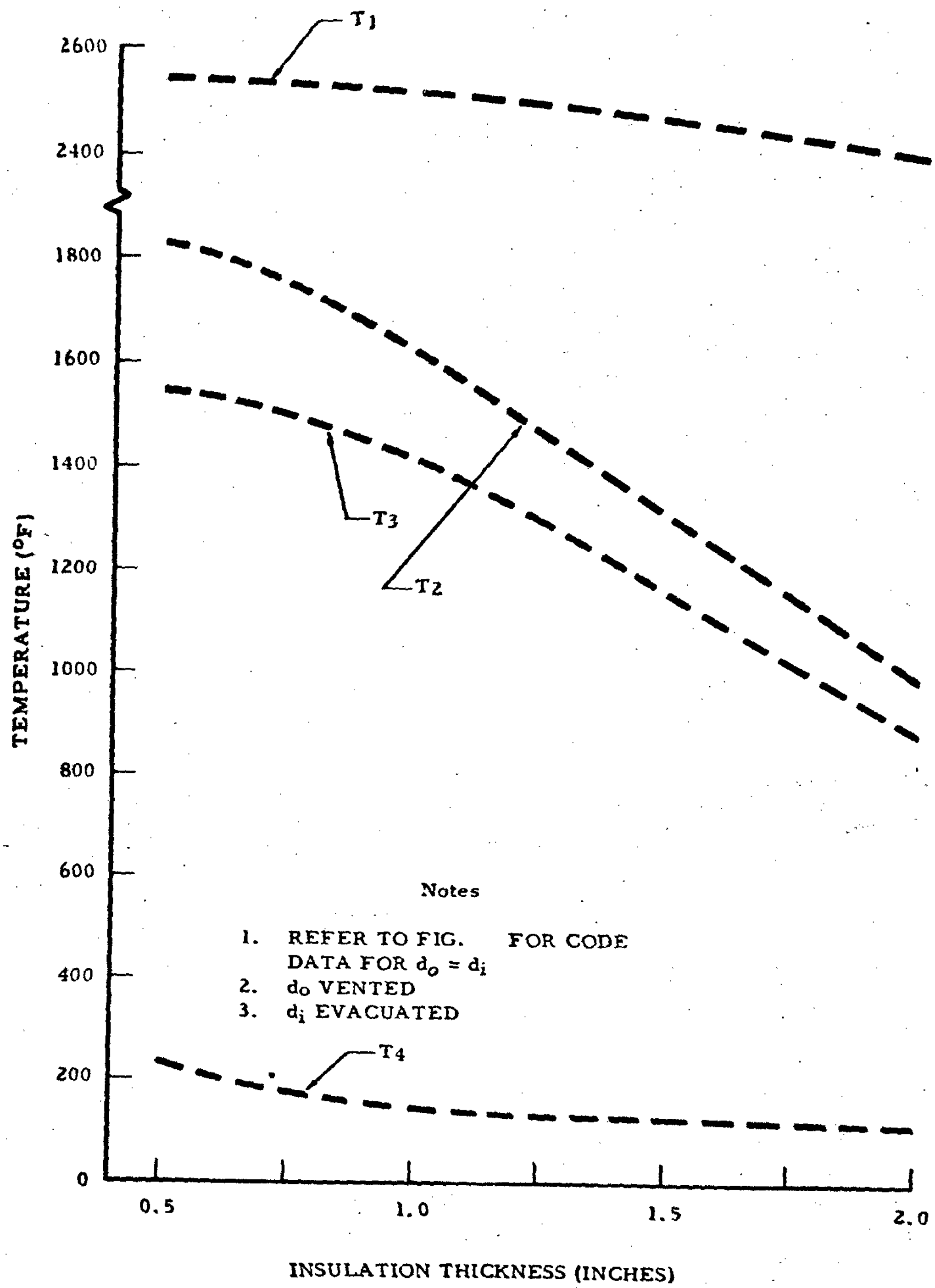


Figure 30. Temperature Versus Insulation Thickness, Vented and Evacuated Walls

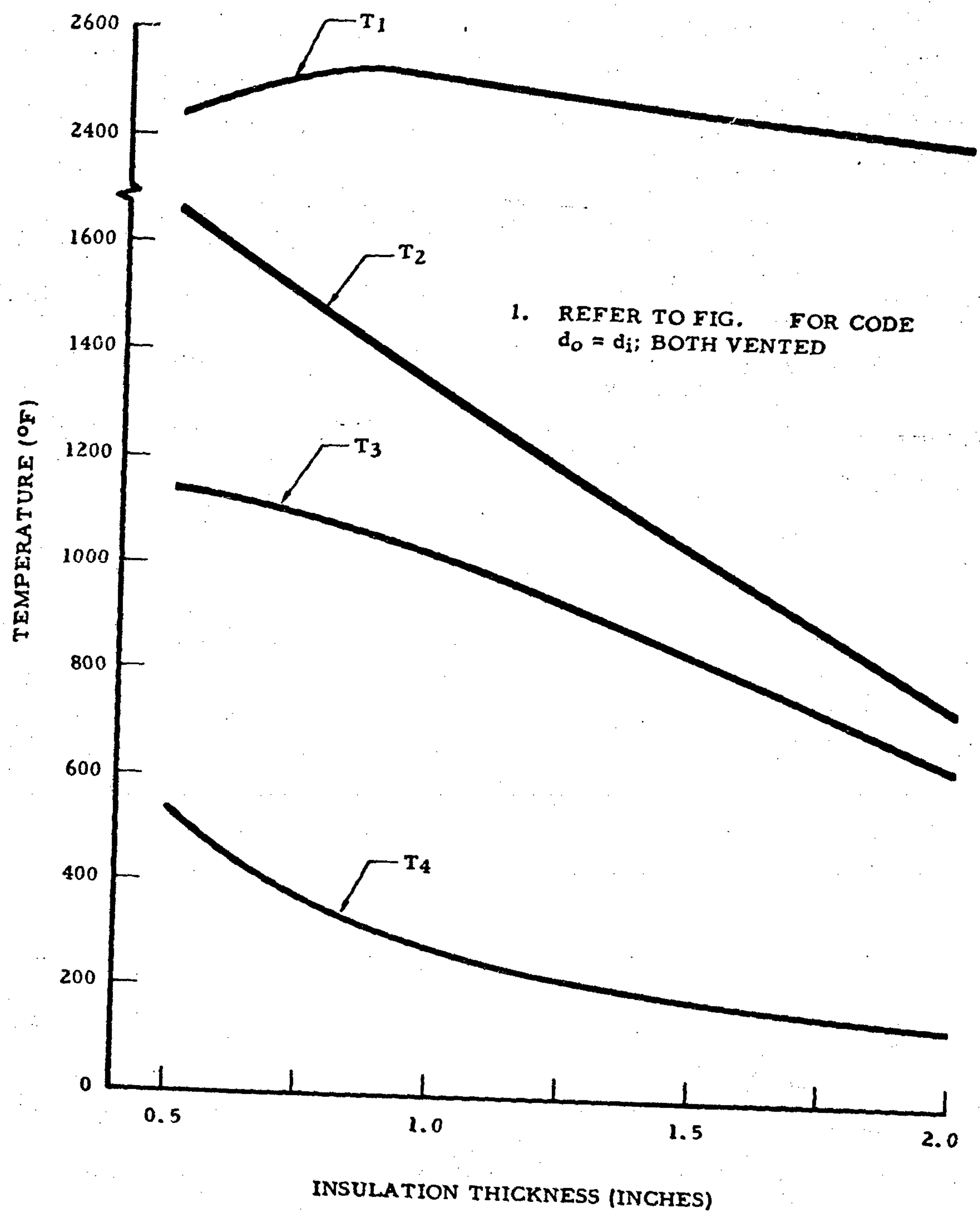


Figure 31. Temperature Versus Insulation Thickness, Vented Walls

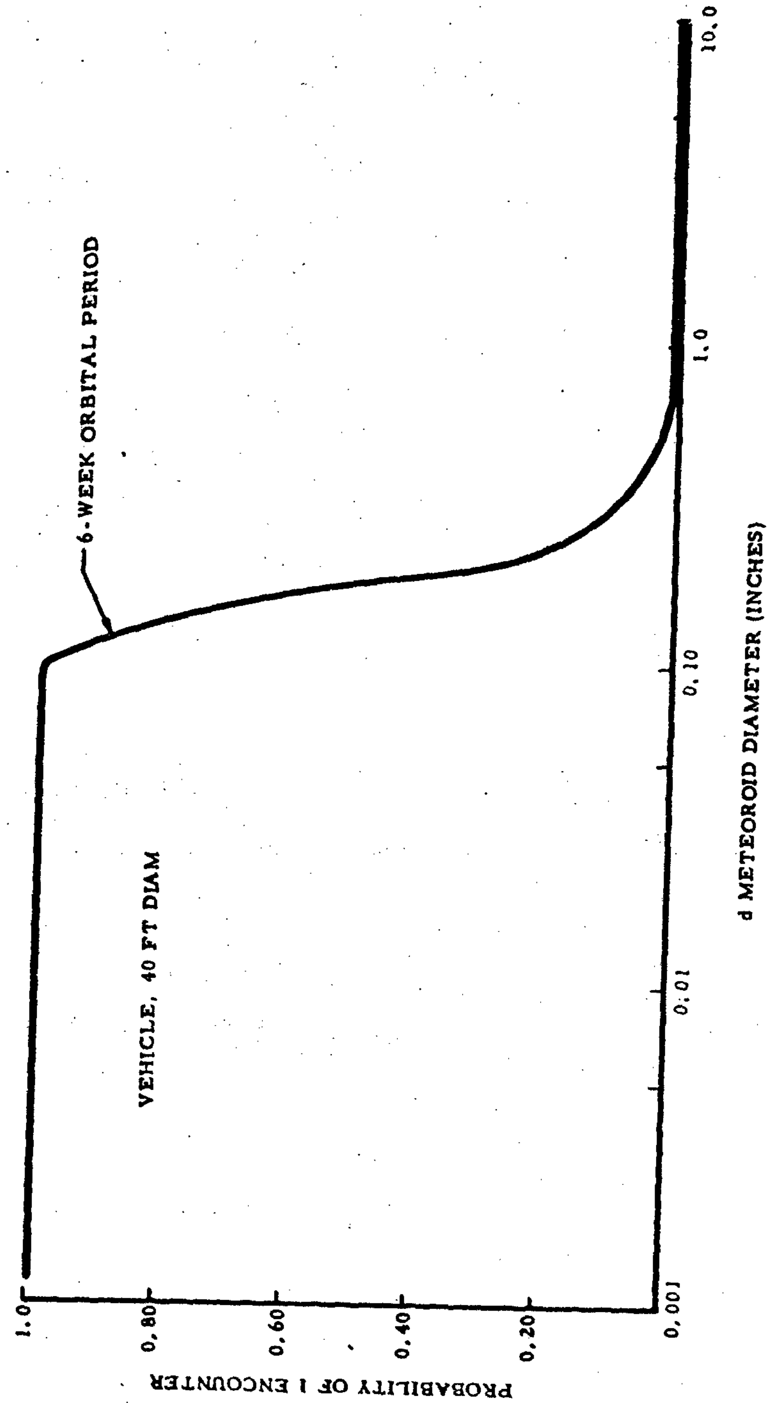


Figure 32. Probability of Meteoroid Encounter Versus Meteoroid Size

Figure 33 shows the required shield thickness needed to ensure zero puncture at various probability levels. It can be concluded that for a cabin area wall thickness of 0.020 aluminum, which is otherwise a structural requirement, a 0.96 probability of not experiencing a puncture can be achieved.

These two figures are representative of an average integrated area of 1260 square feet, representative of the selected vehicle design, exposed to meteoroidal material for a period of 6 weeks.

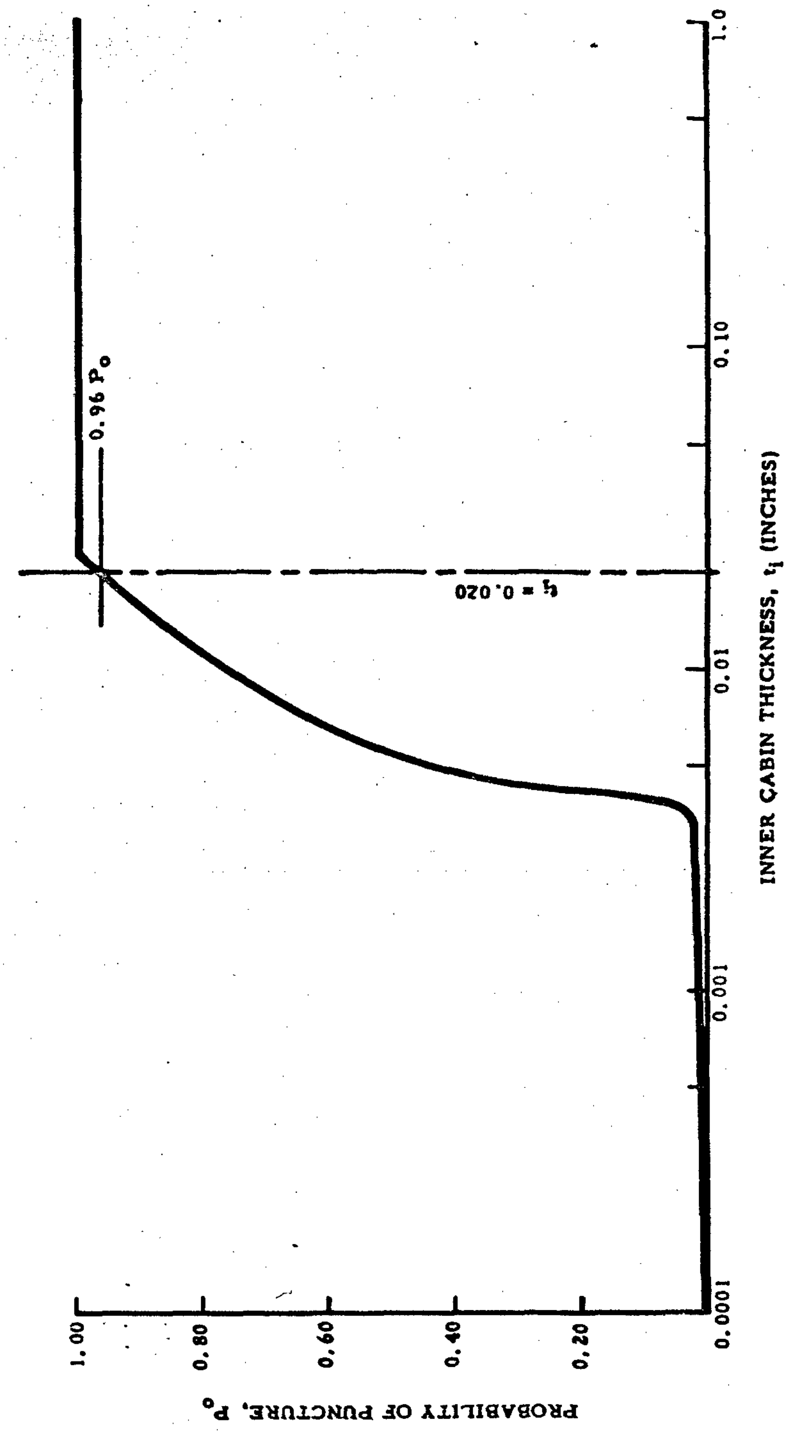


Figure 33. Probability of Survival Versus Required Innercabin Shell Thickness

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2. "NASA Congressional Hearings for Fiscal Year 1961", Parts I and II, March and June 1960, respectively.

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7. ASD TR-61-20, Volume III (SR 178), "Primary Vehicle Global Surveillance System", May 1961.

Aeronautical Systems Division, Dir/Aeromechanics, Flight Accessories Lab, Wright-Patterson AFB, Ohio. Rpt No. ASD-TR-61-240, Pt II, Vol II. ENVIRONMENTAL CONTROL SYSTEMS SELECTION FOR MANNED SPACE VEHICLES: Appendix I -- Missions, Vehicles, and Equipment (U). Final report, Oct 62, 76p. incl illus., tables, 7 refs.

Secret Report

Four versions of manned orbital reentry basepoint vehicles are developed for the purpose of providing reference points for determination of the thermal and atmospheric control requirements of realistic vehicles. Two of the vehicles (i.e., Vehicles 1A and 1B) were developed in Phase I of this study series (ASD Technical Report 61-240, Pt I) and will only be summarized here. The remaining two vehicles (Ballistic Reentry and Lenticular Reentry) are presented in greater detail in this report.

(over)

UNCLASSIFIED

1. Control systems
2. Spaceships
3. Integration & Optimization
4. Environmental control system requirements

I. AFSC Project 6146 Task 614609
II. Contract No. AF 33(616)-8323
III. North American Aviation, Inc., Los Angeles, Calif.
IV. R.J. Oberto
V. NA-62-507
VI. Not aval fr OTS
VII. In ASTIA collection

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In addition to the development of specific vehicles, general data have been compiled on the more important aspects of manned space vehicle design, (i.e., flight vehicle power, structures, effects of meteoroids, mission equipment, and examination of these general data for environmental requirements.

Unclassified Abstract

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Unclassified Abstract

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DEPARTMENT OF THE AIR FORCE
HEADQUARTERS AIR FORCE MATERIEL COMMAND
WRIGHT-PATTERSON AIR FORCE BASE, OHIO

Recd 4/16/2001

APR 12 2001

MEMORANDUM FOR DTIC/OCQ (ZENA ROGERS)
8725 JOHN J. KINGMAN ROAD, SUITE 0944
FORT BELVOIR VA 22060-6218

FROM: AFMC CSO/SCOC
4225 Logistics Avenue, Room S132
Wright-Patterson AFB OH 45433-5714

SUBJECT: Technical Reports Cleared for Public Release (Case AFMC 00-265)

1. The following reports listed in the attached HQ AFMC/PAX Memo, 28 Dec 00, para 1.a., b., and c. were reviewed and cleared for public release in accordance with AFI 35-101, 1 Dec 99, *Public Affairs Policies and Procedures*, Chapter 15.

- AD 330051
- AD 333266
- AD B972544

2. Please direct further questions to Lezora U. Nobles, AFMC CSO/SCOC, DSN 787-8583.

Lezora U. Nobles

LEZORA U. NOBLES
AFMC STINFO Assistant
Directorate of Communications and Information

Attachment:
HQ AFMC/PAX Memo, 28 Dec 00, w/1 Atch